NASA-CR-197203

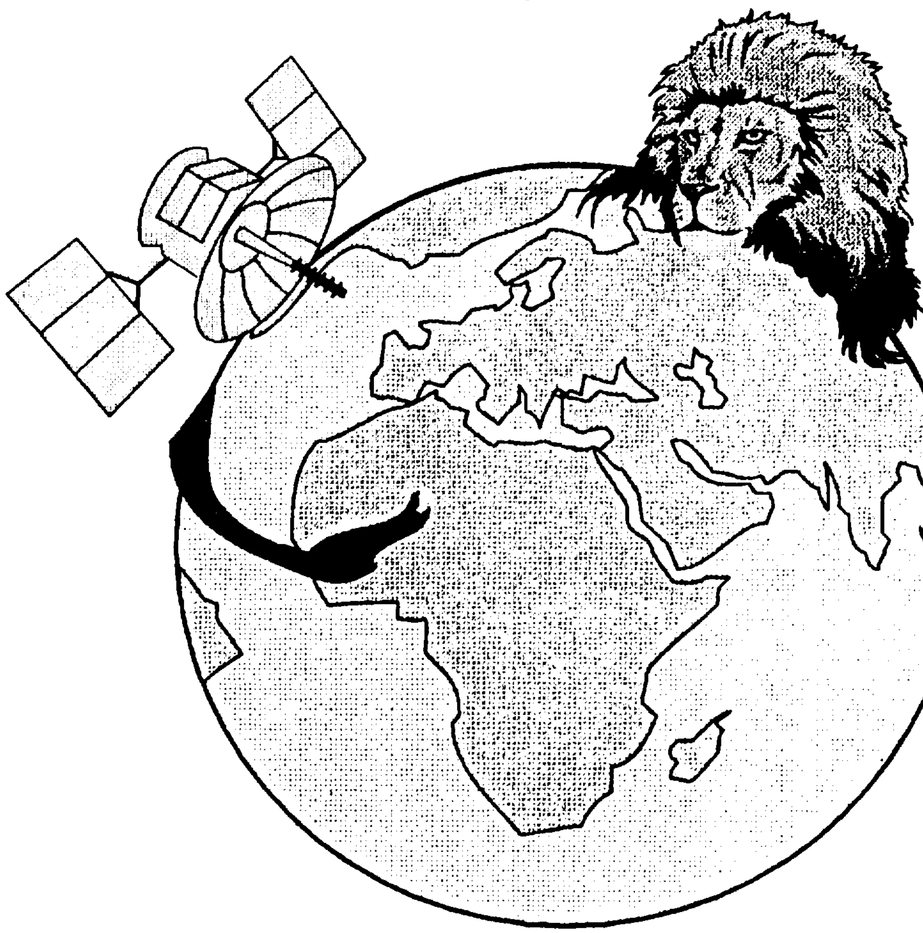
# Minuteman II Launched Small Satellite

## Final Report

NASw-4435

RFP # ASE274L.0194

May 6, 1994





**Submitted to:**
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# Abstract

The goal of LEOSat Industries' Spring 1994 project was to design a small satellite that has a strong technology demonstration or scientific justification and incorporates a high level of student involvement. The satellite is to be launched into low earth orbit by the converted Minuteman II satellite launcher designed by Minotaur Designs, Inc. in 1993. The launch vehicle shroud was modified to a height of 90 inches, a diameter of 48 inches in at the bottom and 35 inches at the top for a total volume of 85 cubic feet. The maximum allowable mass of the payload is about 1,100 lb., depending on the launch site, orbit altitude, and inclination. The satellite designed by LEOSat Industries is TerraSat, a remote-sensing satellite that will provide information for use in space-based Earth studies. It will consist of infrared and ultraviolet/visible sensors similar to the SDI-developed sensors being tested on Clementine. The sensors will be mounted on the Defense Systems, Inc. Standard Satellite-1 spacecraft bus. LEOSat has planned for two satellites orbit the Earth with trajectories similar to that of LANDSAT 5. The semi-major axis is 7,080 kilometers, the eccentricity is 0, and the inclination is 98.2 degrees. The estimated mass of TerraSat is 145 kilograms and the estimated volume is 1.8 cubic meters. The estimated cost of TerraSat is $13.7 million. The projected length of time from assembly of the sensors to launch of the spacecraft is 13 months.

## EXECUTIVE SUMMARY

### Introduction

As a result of the Strategic Arms Reduction Talks (START), 450 Minuteman II (MMII) ICBMs will be dismantled. In 1993, Minotaur Designs, Inc. (MDI) recognized that these excess missiles could be used to accomplish goals other than nuclear deterrence and worldwide destruction. In response to the need for a cost-effective satellite launch system, MDI designed a converted MMII satellite launcher. Consequently, the opportunity to design small low Earth orbit (LEO) satellites to make use of this new launcher arises. LEOSat Industries has completed a preliminary design of TerraSat, an Earth-sensing satellite that utilizes new miniaturized sensors currently being flown on the Clementine spacecraft. The data provided by TerraSat can be processed into images by graduate and undergraduate students and can also be used to teach secondary students Earth sciences such as geology and geography. TerraSat will also function as a backup satellite for LANDSAT 5 which is operating well beyond its three year design life. Since LANDSAT 6 failed to reach orbit, TerraSat can bridge a possible gap in continuous data between a failure of LANDSAT 5 and the launch of LANDSAT 7.

### Assumptions and Requirements

The following requirements were placed upon LEOSat by the initial mission statement:

- LEOSat must develop several candidate satellites and choose one from among them.
- The satellite will be launched by the Minuteman II booster designed by MDI in 1993.
- The satellite must have a strong technology demonstration or scientific justification.
- The satellite must have a high level of student involvement at multiple levels.

In addition to these requirements, the following assumptions were made:

- Any available Star injection stage may be used.
- The failure of LANDSAT 5 is imminent.
- The Clementine sensors will be available for commercial use.

**Selection of Satellite**

LEOSat began this project by considering four small satellite projects. Two have been chosen for primary and secondary projects and two have been ruled out. The primary project, TerraSat, is an Earth-sensing satellite and a detailed design of this satellite is required by the mission statement. The secondary project, COBE Jr., is a follow-on to the scientific mission performed by the Cosmic Background Explorer (COBE) and only a preliminary design is required for COBE Jr. The other two candidates, the SOS satellite and the Crystal Growth Platform, were not selected as design projects .

**Spacecraft Sensors**

TerraSat will consist of four sensors similar to those currently being flown on the Clementine spacecraft. These sensors are small and lightweight and require much less power than the current sensors being used for remote sensing. The four sensors are the Short-Wave IR Camera, Long-Wave IR Camera, Low-Resolution UV/Visible Camera and High-Resolution Visible Camera. The sensors are capable of taking 20 images per second. Three of the sensors utilize filter wheels which allow them to measure a variety of wavelengths. Since the time to change filters and dampen jitter is approximately 200-250 milliseconds, they can provide multispectral images. These sensors were mounted to the Clementine spacecraft through the use of an optical plate which did not flex much with temperature or vibration. This meant that the sensors were not affected by vibration of the spacecraft bus. TerraSat includes the use of the same optical plate, so that the images produced will not be affected by the spacecraft bus. The attributes of the sensors are listed in Table 1.

**Table 1. Sensor Attributes**

| Sensor | Mass (kg) | Power (W) | Size (cm) | Wavelengths (μm) | Resolution (m) |
|---|---|---|---|---|---|
| Short-Wave IR | 1.6 | 30 | 36.8 x 11.2 x 11.7 | 1.1 - 2.78 | 130 |
| Long-Wave IR | 1.65 | 30 | 39.1 x 14.7 x 14.7 | 8.0 - 9.5 | 50 |
| Low-Res UV/Visible | 0.5 | 6 | 15.5 x 11.7 x 10.4 | 0.4 - 1.0 | 90 |
| High-Res UV/Visible | 1.25 | 12 | 36.8 x 9.1 x 17.8 | 0.4 - 0.75 | 12 |

## Mechanical Support Structure

TerraSat will connect directly to the payload support bulkhead, shown in Figure 1, with the use of the flexible Marmon clamp (designed by MDI and modified by LEOSat), shown in Figure 2, and four truss elements shown in Figure 3. The support structure is capable of cradling up to a 1500 pound payload, and can easily withstand the most extreme expected launch environment of 9 g's axial and 3.75 g's tangential acceleration.

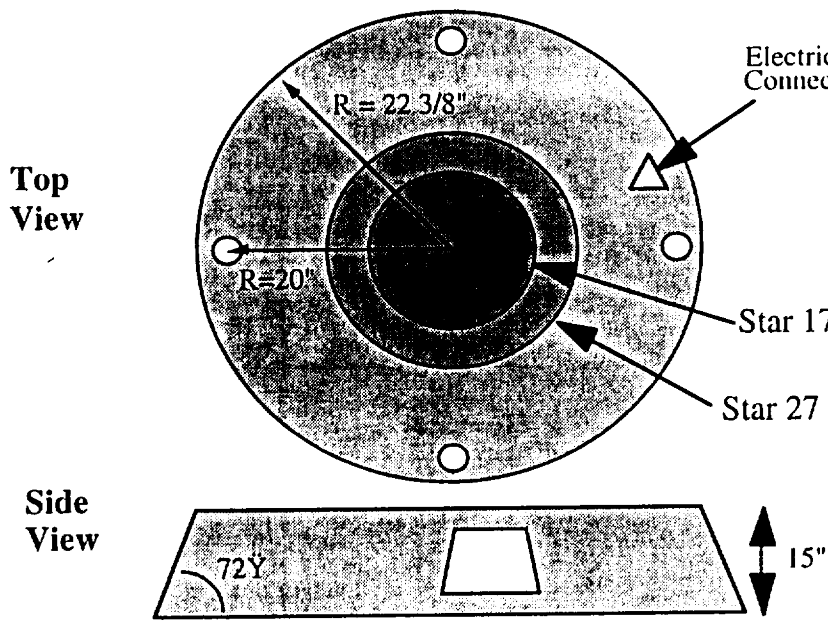


**Figure 1. Payload Support Bulkhead**

The PSB used on TerraSat was designed by MDI and is designed to mount a variety of load bearing structures and support the payload electrical interface. Each circle represents a connection point which consists of 5 slots. The trusses are bolted through the slots. Each connection point has a total tension pull-out capability of 15,000 lbs. The triangular object represents the electrical interface connector which is actually circular to avoid stress concentrations.

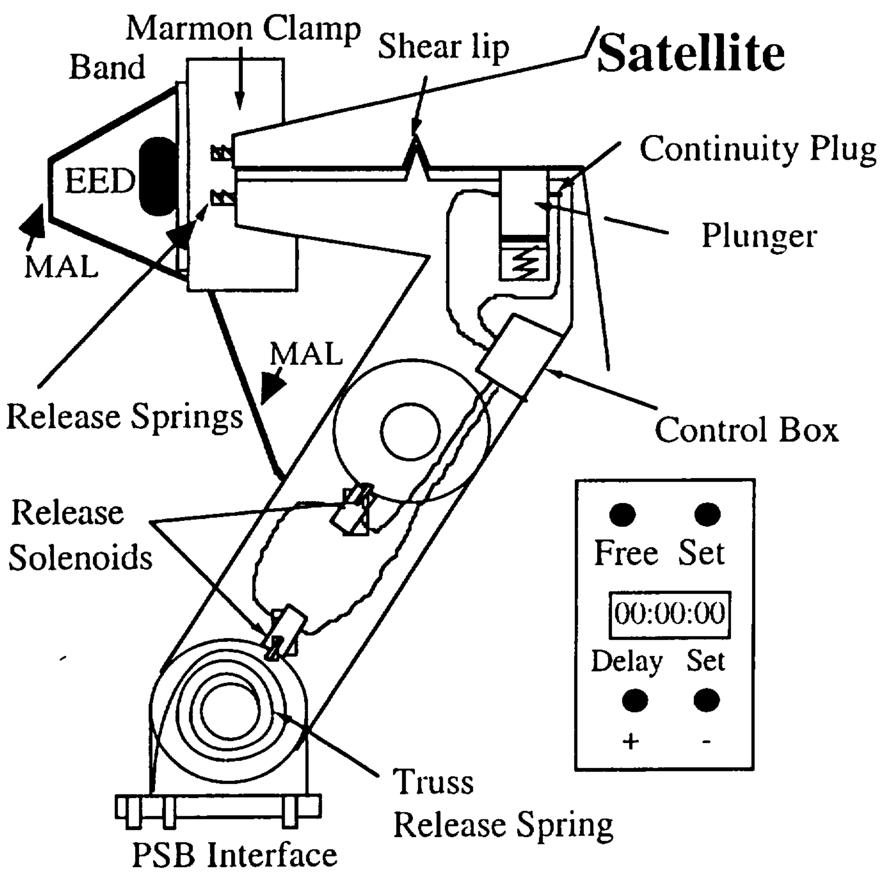


**Figure 2. Payload Attach Fitting (Truss Elements)**

LEOSat has made one design change to the MDI Marmon clamp, shown in Figure 3. In order to decrease the possibility of creating more space debris, LEOSat has added a 100 lb-test monofilament line to the Marmon clamp to hold the clamp together and avoid having pieces of the clamp flying into space as debris. The lines will be held on the clamp with high temperature epoxy, and will be placed in several locations on the clamp for redundancy. The weight of this monofilament line is negligible. All other aspects of the Marmon clamp will be the same as the design by MDI.

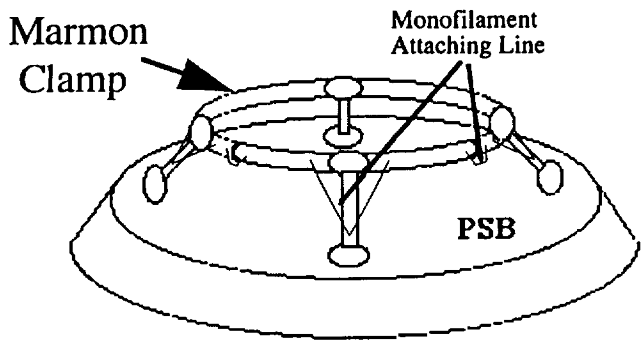


**Figure 3. PSB with modified Marmon clamp**

## Orbit Analysis

Since TerraSat will provide data similar to that of the LANDSAT series of satellites, LEOSat has decided to use similar orbital elements. LANDSAT 5 is in a circular orbit, has an inclination of 98.22°, and has a semi-major axis of 7083 km (altitude = 705 km). Another requirement was that the satellite be in a sun-synchronous orbit and be out of the shadow of the Earth. To determine which injection stage will be needed to achieve similar orbital elements, two TK! Solver routines were used. The chosen orbit has an inclination of 98.2 degrees, a semi-major axis of 7080 km, and an altitude of 702 km.. To achieve this inclination, the satellite must be launched from Vandenberg AFB and a Star 10 injection stage must be used. The chosen orbit resulted in a final mass for TerraSat of 145 kg. This extra 18 kg of mass may be accounted for by error margin, ballast mass, and the extra cryocoolers needed for the two infrared sensors. The analysis results in a sun-synchronous orbit that is out of the Earth's shadow and is very similar to the orbit of LANDSAT 5.

## Guidance, Navigation, and Control

Since TerraSat is an Earth-sensing satellite, it must remain pointed at the Earth at all times. As a result, the DSI bus housing the TerraSat payload will be nadir pointing and three-axis stabilized, with a pointing error of 0.03 degrees. To monitor the small angular rates of change, a combination of two star trackers, a three-axis magnetometer, and a horizon scanner will be used. In order to maintain TerraSat's altitude, small frequent maneuvers will probably be required to

counteract orbital perturbations due to such effects as atmospheric drag and solar radiation pressure. LEOSat will use two reaction wheels and XYZ torque coils to perform such maneuvers.

**Communications**

TerraSat will require a transmitter and a receiver to communicate with the ground station. Since uplink commands to TerraSat will require a modulation rate of no more than 9.6 kilobits/sec, a UHF-band command uplink should be adequate. However, due to the large amount of data to be transmitted to the ground station by the satellite, an S-band downlink with a modulation rate of at least 1 Megabit/sec will be required. To provide the capability of continuous imaging without the requirement of numerous ground stations, 100 Megabytes of on-board memory will be required.

**Power/Thermal**

The total power required by the sensors is approximately 80 W. This power will be supplied by 3 sets of deployable solar panels, each set providing 30 W for a total available power of 90 W. Body mounted solar panels will also be used for redundancy. The body mounted solar panels are capable of providing up to 20 W of orbit average power.

Two of the four sensors, the Short-Wave and Long-Wave IR sensors, have their own Integral Sterling type Ricor K506B cryocoolers. The coolers have an average mean-time-to-failure of 4000 hours and will maintain the sensors at a temperature between 50 K and 77 K. A total of six of these coolers will be added on board for a total design operational time of three years. The other two sensors will be kept between 253 K and 283 K by using the spacecraft bus as a heat sink.

**Ground Operations**

LEOSat has analyzed the feasibility of using existing LANDSAT ground stations to receive the data transmitted from TerraSat and has found this approach unfeasible. However, if LEOSat Industries can build a ground station modeled on the current EOSAT ground stations , it can begin an operation similar to that of EOSAT. It is estimated that construction of such a ground station will cost approximately $1 million. Personnel costs and operational costs for this ground station will be

approximately $0.5 million per year each. The data received by the ground station will be stored on high density tapes by a VAX-based computer. These tapes will be shipped directly to LEOSat headquarters in Austin, Texas, where they will undergo their first evaluation for quality (line drops, pixel noise, etc.) and cloud cover and will then be distributed to image processing centers across the United States.

**Student Involvement**

Graduate students will utilize current techniques to process the images and data as well as develop new processing techniques. These students will also write lesson plans and make presentations to secondary schools. The total cost for graduate student involvement is estimated at $660,000. Elementary and high school students will use the images to study geography, geology, and other Earth sciences. A direct mail-out advertising the educational materials available from LEOSat industries and the teacher-outreach offices at the NASA centers around the country will cost approximately $96,000. Undergraduate students will also have access to the images through the Internet distribution system. Distributing the processed images to the public on the Internet system will cost $73,000 for the first year and $29,000 for each subsequent year. The total cost of ground operations for the first year is estimated at $2.7 million and $1.2 million for each subsequent year. LEOSat has also considered the possibility of using the LASP facility at the University of Colorado, or building a similar facility near the University of Texas at Austin to enable undergraduate and graduate students to participate in the satellite monitoring and data gathering phases of the TerraSat project. However, LEOSat has not determined the cost or the willingness of either university to participate in such a program.

**Satellite Bus Selection**

LEOSat Industries had to select a commercially available satellite bus that would be compatible with the MMII launch vehicle and had a maximum allowable mass of 1100 lb including the sensors. LEOSat selected the Standard Satellite-1 made by Defense Systems, Inc., an 8-sided, modular satellite developed for medium size payloads that are designed to operate in low Earth orbit. The SS-1 is 30 inches in diameter and has a core module height of 16.5 inches. The payload is

housed in a 16 inch high payload module. The maximum allowable payload mass is 400 lb. and the maximum total mass of the bus and payload is 625 lb. The bus provides up to 150 Watts of power and may be 3-axis or spin stabilized. Figure 4 shows a simple diagram of the DSI SS-1 satellite bus.

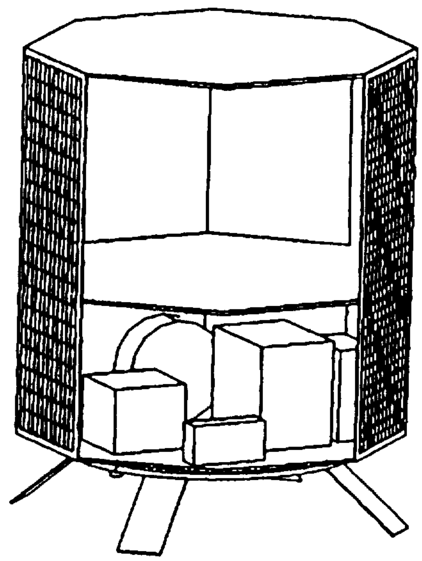

**Figure 4. DSI SS-1 Satellite Bus**

**Mass, Volume, Cost, and Timeline Estimate**

The estimated mass of TerraSat is 145 kg. The volume has been estimated at 1.8 $m^3$. The cost of each satellite is approximately $13.7 million. The total estimated time for construction, integration and launch of TerraSat is 13 months.

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# 1.0 INTRODUCTION

## 1.1 Project Background

The end of the Cold War between the United States and the former Soviet Union brought not only a relaxation of international tension, but also a corresponding decrease in the number of intercontinental ballistic missiles (ICBMs). As a result of the Strategic Arms Reduction Talks (START), 450 Minuteman II (MMII) ICBMs will be dismantled. In 1993, Minotaur Designs, Inc. (MDI) recognized that these excess missiles could be used to accomplish goals other than nuclear deterrence and worldwide destruction. In response to the need for a cost-effective satellite launch system, MDI designed a converted MMII satellite launcher. Consequently, the opportunity to design small low Earth orbit (LEO) satellites to make use of this new launcher arises.

Small satellites may be used for many of the same purposes as larger satellites, although the payload size and mass, and consequently the number of scientific experiments that can be included in the payload, are limited. But lower satellite mass leads to lower launch costs, while including fewer experiments results in both lower costs and shorter delivery times. As government funding becomes even less available in the next few years, only projects with low costs and high scientific or technological justification that can also excite the general public about the opportunities of space exploration will obtain funding.

## 1.2 Objectives

LEOSat Industries has completed a preliminary design of two small Earth satellites to be boosted into low Earth orbit by the Minotaur MMII converted military booster specified above. The two satellites are designated "primary" and "secondary", with the "primary" design being that which best fits the criteria presented in the Request For Proposal (RFP). These criteria include those constraints imposed by the MMII launch vehicle (i.e. trajectory, mass, and altitude limitations, shown in Appendix B of the Conceptual Design Review), strong scientific or technology demonstration justification, cost, and potential for involvement of students at the graduate, undergraduate, and secondary school levels.

## 1.3 Selection of Project

LEOSat began this project by considering four small satellite projects. Two were chosen for primary and secondary projects and two were ruled out. The primary project, TerraSat, is an Earth-sensing satellite and is discussed in Section 2. The secondary project, COBE Jr., is a follow-on to the scientific mission performed by the Cosmic Background Explorer (COBE) and is discussed in Appendix C. The other two candidates, the SOS satellite and the Crystal Growth Platform, were not selected as design projects but are discussed briefly in Appendices D and E.

## 1.4 Assumptions and Requirements

The following requirements were placed upon LEOSat by the initial mission statement:

- Must develop several candidate satellites and choose one from among them
- Satellite will be launched by the Minuteman II designed by MDI in Fall 1993
- Satellite must have a strong technology demonstration or scientific justification
- Satellite must have a high level of student involvement at multiple levels

In addition to these requirements, the following assumptions were made:

- Any available Star injection stage can be used.
- The failure of LANDSAT 5 is imminent.
- The Clementine sensors will be available for commercial use.

## 2.0 PRIMARY DESIGN PROJECT - TerraSat

### 2.1 Background

Throughout history, mankind has struggled to understand the many changes which take place in the world. Space-based remote sensing of the Earth has become an important tool in increasing this understanding. The LANDSAT series of satellites have been providing information about the Earth for over twenty years with a continuous flow of data that is required to study global trends. Fire hazards to national, state and private parks are being monitored using the data from LANDSAT. Urban growth has been monitored and the environmental effects of this urban growth on the surrounding rural areas determined. LANDSAT data has been used to locate water sources and map shorelines. The total acreage of the shrinking wetlands and rainforests is being monitored. Private companies are using the LANDSAT data to determine the economic feasibility of mineral resource sites and develop environmentally-friendly plans to develop these resources. But this flow of useful data may be stopped.

LANDSAT 5 was launched in 1984 with a design life of 3 years and has been operating for 10 years. Its replacement, LANDSAT 6, failed to reach orbit in October 1993. If LANDSAT 5 fails, the resulting gap in data could jeopardize the work that is being done to study the Earth. Although the development of LANDSAT 7 has been accelerated, it will still be at least 4 years before it can be launched. For this reason, LEOSat has proposed TerraSat, a small remote sensing satellite which can be launched into low Earth orbit by a converted Minuteman II missile in approximately one year. This satellite would provide remote sensing data similar to that being gathered by LANDSAT 5 as well as testing for new miniaturized sensors in an Earth-sensing application.

### 2.2 Project Summary

The purpose of TerraSat is to measure the reflected energy of the Earth's surface. Every object in the universe radiates and reflects a certain amount of energy at a particular wavelength. By measuring the reflected energy of the Earth at several different wavelengths, objects can be identified from space. Most reflected energy has a wavelength within the visible or infrared portion

of the spectrum. For example, a plant's chlorophyll concentration can be measured in the 0.45-0.52 micron region of the spectrum while the vegetation density can be measured in the 0.76-0.90 micron region of the spectrum. TerraSat will take measurements in the ultraviolet, visible and infrared portions of the spectrum. These measurements will enable differentiation between vegetation and soil, measurement of the health and plant density of vegetation, and mapping of geological formations. The satellite will consist of four sensors mounted on a commercially available satellite bus that provides all the necessary subsystems (see Section 4).

## 2.3 Mission Scenarios

### 2.3.1 Primary Scenario

Figure 5.4-1 in Section 5.4 shows the estimated timeline for the construction of TerraSat. The sensors will be built in approximately two months and will be transported to DSI during a two week time period . The next two and a half months will be used to integrate the sensors into the payload bus. Once it has been integrated, LEOSat will allow four weeks for the satellite to be shipped to Vandenberg AFB. At Vandenberg, the satellite will undergo a six month integration into the MMII launch vehicle. The schedule for the month of pre-launch testing of the spacecraft will be detailed as the complete design of the satellite develops. The construction and launch preparation of the satellite will take approximately 13 months. The satellite is then designed to have a minimum three year life on orbit.

TerraSat will do selective imaging until such time as LANDSAT 5 fails. When LANDSAT 5 fails, TerraSat will be switched to constant imaging to provide nearly the same data that the failed satellite had been providing. When in continuous imaging mode, TerraSat will be transmitting much more data to ground control and more personnel may be required for image processing. From the time LANDSAT 5 fails until the time LANDSAT 7 is launched and fully operational, the image processing facilities currently being used by EOS will not be in use and it is possible that LEOSat could make use of these facilities.

### 2.3.2 Back-up Scenario

Three satellites will be built in the first round of construction. Two of these satellites will be launched into orbit following the schedule discussed in Section 2.3.1 The third satellite will be used as a back-up. In case of failure of one of the primary satellites or any accidents during launch, a third satellite will be ready to be integrated into a launcher and launched, thus helping to ensure success of the mission. If LEOSat decides that more than two satellites are necessary to provide complete and timely sensing coverage, the additional satellites will be built in the second round of construction. This will ensure that any errors or misdesigns can be corrected before more money is wasted on additional satellites.

## 2.4 Student Involvement

The design requirements for the Minuteman II small satellite project include the requirement for a high level of student involvement. For the primary design, the student involvement would occur at all levels of the education system. Undergraduate and graduate students would utilize current techniques to process the images and data as well as develop new processing techniques. Elementary and high school students would use the images produced by TerraSat to study geography, geology, and other Earth sciences. LEOSat hopes that the data will also begin to excite students to the possibilities and uses of space and satellites and will encourage them to consider careers in science and engineering. Undergraduate students will also have access to the images to get an introduction to the uses of satellites and the methods of image processing.

LEOSat has also considered the possibility of having undergraduate and graduate students participate in the satellite monitoring and data gathering phases of the TerraSat project. Using the LASP facility at the University of Colorado, or building a similar facility near the University of Texas at Austin would enable students to control the satellite and its imaging resources and learn about mission design and control. LEOSat has not researched the cost of such a program or the willingness or ability of the University of Colorado or the University of Texas to particpate. These two issues would be the deciding factors in whether or not this phase of the project is feasible.

## 3.0 SUBSYSTEMS ANALYSIS

### 3.1 Sensors

TerraSat will consist of four sensors which will be very similar to those currently being flown on the Clementine mission. The sensors will be mounted in a commercially available satellite bus. These sensors are small and lightweight and require much less power than the current sensors being used for remote sensing. Each sensor will monitor a different portion of the spectrum and will provide the same coverage as six of the seven bands covered by LANDSAT 5. The four sensors are the Short-Wave IR Camera, Long-Wave IR (LWIR) Camera, Low-Resolution UV/Visible Camera and High-Resolution Visible Camera. The sensors are capable of taking 20 images per second. Three of the sensors utilize filter wheels which allow them to measure a variety of wavelengths. Since the time to change filters and dampen jitter is approximately 200-250 milliseconds, the sensors can provide multispectral images. These sensors were mounted to the Clementine spacecraft through the use of an optical plate which did not flex much with temperature or vibration. This meant that the sensors were not affected by vibration of the spacecraft bus. TerraSat includes the use of the same optical plate, so that the images produced will not be affected by the spacecraft bus.

#### 3.1.1 Short-Wave IR Camera

The specifications for the Short-Wave IR Camera are listed in Table 3.1-1. This sensor is a cooled video camera with a ground resolution of approximately 130 meters. The Short-Wave IR Camera also has six filters which provide coverage at 1.10, 1.25, 1.50, 2.0, 2.60, and 2.78 microns. It has a mechanical cooler which allows operation at a low temperature. The estimated cost of this sensor is $400,000. The mass, power, and cost include the mechanical cooler.

#### 3.1.2 Long-Wave IR Camera

The specifications for the Long-Wave IR Camera are listed in Table 3.1-2. This sensor is also a cooled video camera with a ground resolution of approximately 50 meters. The Long-Wave IR Camera does not have a filter wheel. It takes wide bandwidth measurements between 8.0 and 9.5 microns and has a mechanical cooler which allows operation at a low temperature. The estimated cost of this sensor is $400,000. The mass, power, and cost include the mechanical cooler.

**Table 3.1-1. Specifications for Short-Wave IR Camera**

| | |
|---|---|
| Mass (grams) | 1600 |
| Size (cm) | 36.8 x 11.2 x 11.7 |
| Electrical Power (Watts) | 30 |
| Wavelength (microns) | 1.1 - 2.78 |
| Field of View (degrees) | 5.6 x 5.6 |
| Pixel Format | 256 x 256 |
| Images per Second | 10 |
| Focal Plane Array | InSb |
| Filter Wheel Positions | 6 positions |

**Table 3.1-2. Specifications for Long-Wave IR Camera**

| | |
|---|---|
| Mass (grams) | 1650 |
| Size (cm) | 15 x 15 x 40 |
| Electrical Power (Watts) | 30 |
| Wavelength (microns) | 8.0 - 9.5 |
| Field of View (degrees) | 1.0 x 1.0 |
| Pixel Format | 128 x 128 |
| Images per Second | 10 |
| Focal Plane Array | HgCdTe |
| Filter Wheel Positions | Fixed |

### 3.1.3 Low-Resolution UV/Visible Camera

The specifications for the Low-Resolution UV/Visible Camera are listed in Table 3.1-3. The Low-Resolution UV/Visible sensor is a charge-coupled device video camera which has a pixel resolution of approximately 90 meters if placed in an orbit similar to LANDSAT 5. Six bandpasses can be selected through the use of filter wheels. These bandpass filters are 0.400, 0.415, 0.750, 0.900, 0.950, and 1.000 microns. The estimated cost of this sensor is $250,000.

**Table 3.1-3. Specifications for Low-Resolution UV/Visible Camera**

| | |
|---|---|
| Mass (grams) | 500 |
| Size (cm) | 15.5 x 11.7 x 10.4 |
| Electrical Power (Watts) | 6 |
| Wavelength (microns) | 0.4 - 1.0 |
| Field of View (degrees) | 4.2 x 5.6 |
| Pixel Format | 384 x 288 |
| Filter Wheel | 6 positions |

### 3.1.4 High-Resolution UV/Visible Camera

The specifications for the High-Resolution UV/Visible Camera are listed in Table 3.1-4. This sensor is a charge-coupled device camera with a ground resolution of approximately 12 meters. The High Resolution UV/Visible Camera also has six filters which provide coverage at 0.400, 0.415, 0.560, 0.650, and 0.750 microns. The estimated cost of this sensor is $1 million.

**Table 3.1-4. Specifications for High-Resolution UV/Visible Camera**

| | |
|---|---|
| Mass (grams) | 1250 |
| Size (cm) | 36.8 x 9.1 x 17.8 |
| Electrical Power (Watts) | 12 |
| Wavelength (microns) | 0.4 - 0.75 |
| Field of View (degrees) | 0.3 x 0.4 |
| Pixel Format | 384 x 288 |
| Images per Second | 20 |
| Focal Plane Array | Si CCD |
| Filter Wheel Positions | 6 positions |

## 3.2 Mechanical Support Structure

Few changes will be made in the support structure designed by MDI in the fall of 1993. The same payload support bulkhead (PSB), tubular trusses, and Marmon clamp designs can be modified to fit the needs of LEOSat and TerraSat. The following information will outline MDI's initial designs and the modifications made by LEOSat.

TerraSat will connect directly to the payload support bulkhead shown in Figure 3.2-1 with use of the flexible Marmon clamp (designed by MDI and modified by LEOSat) shown in Figure 3.2-3 and four truss elements shown in Figure 3.2-2. The support structure is capable of cradling up to a 1500 pound payload, and can easily withstand the most extreme expected launch environment of 9 g's axial and 3.75 g's tangential acceleration.

### 3.2.1 Payload Support Bulkhead

The TerraSat PSB is exactly the same as the MDI design. The PSB is designed to mount a variety of load bearing structures and support the payload electrical interface. Each circle represents a connection point which consists of 5 slots. The trusses are bolted through the slots. Each connection point has a total tension pull-out capability of 15,000 lbs. The triangular object represents the electrical interface connector which is actually circular to avoid stress concentrations. The PSB dimensions are listed in Table 3.2-1 . Material properties are listed in Table 3.2-2. The PSB is shown in Figure 3.2-1.

**Table 3.2-1. PSB dimensions**

| | |
|---|---|
| Height | 15 in. |
| Base Diameter | 52 in. |
| Bulkhead Diameter | 47.5 in. |
| Truss attach hole radius | 20 in. |
| Electrical hole radius | 20 in. |
| Electrical hole off axis | 45 deg. |

**Table 3.2-2. PSB material properties**

| | |
|---|---|
| Material | Aluminum 7075 |
| Finish | Alodine 600 |
| Thickness | 2.0 inches |
| Weight | 25 lb. |
| Truss tension pullout capability | 15,000 lb. per truss |

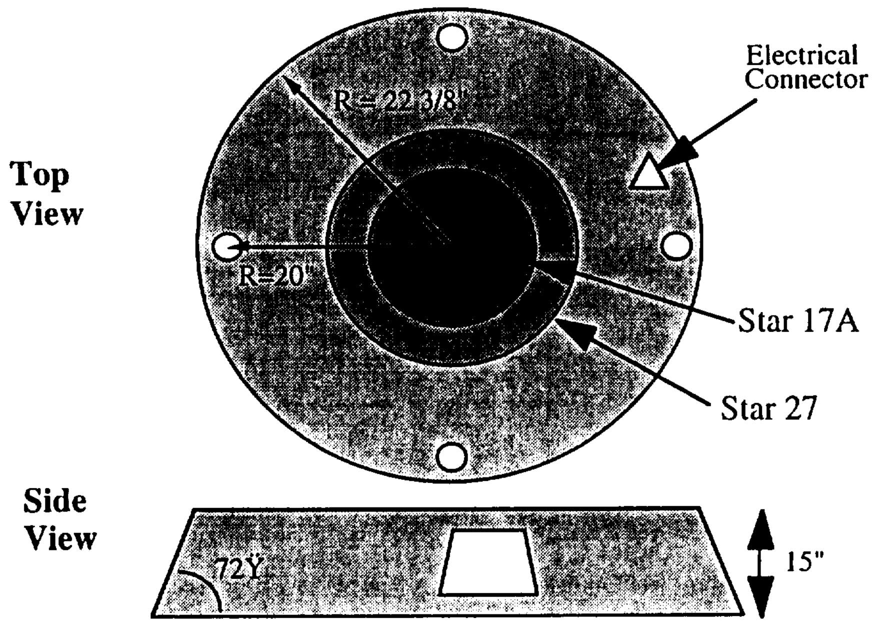


**Figure 3.2-1. Payload Support Bulkhead**

3.2.2 Payload Attach Fitting

The MDI-designed truss shown in Figure 3.2-2, will be used for the TerraSat to attach the Marmon clamp to the PSB. No changes will be made to the MDI design. Table 3.2-3 lists the truss material properties. These trusses will transfer the satellite loads during launch and can be adjusted to the correct size of a 30 inches diameter. Each truss element consists of a rod, a clamping end, and two hinges. These hinges can be set to properly fit TerraSat. The trusses are connected to the payload lip with four wedges and one cinching ring. The Electrical Explosive Device will be used to cut the cinching ring. Springs will eject the wedges radially and springs will eject the payload. Redundant separation initiators and redundant separation signals will be designed into the system.

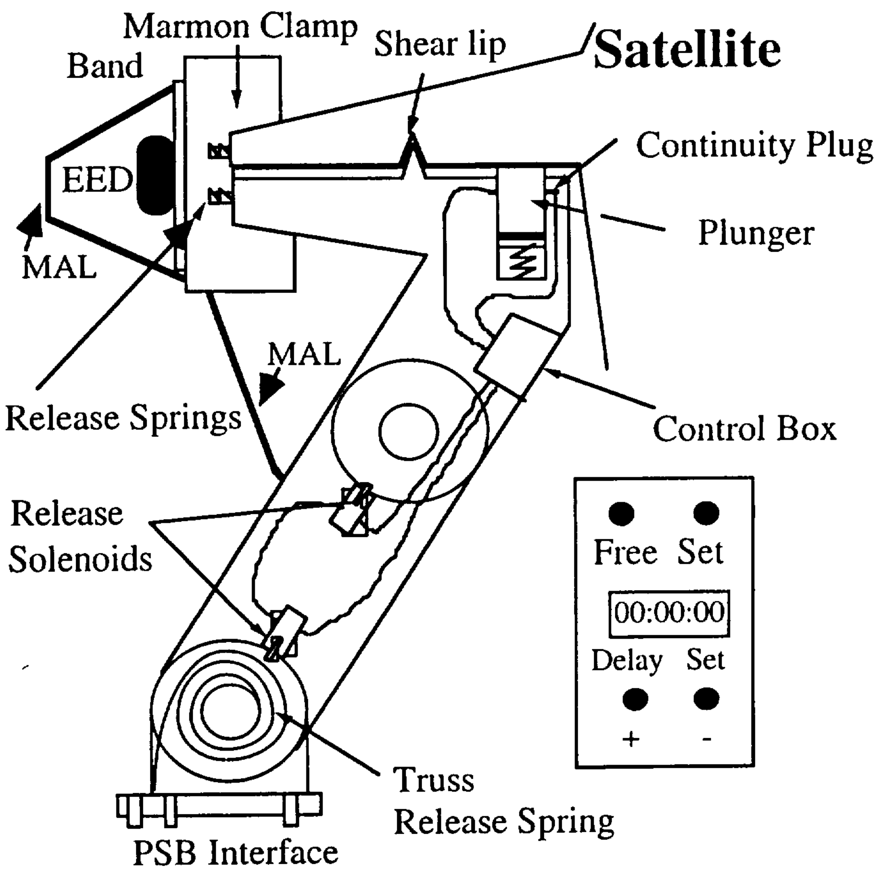


**Figure 3.2-2. Payload Attach Fitting (Truss Elements)**

**Table 3.2-3 Truss material properties**

| | |
|---|---|
| Material | Steel |
| Nominal Length | 32 inches |
| Nominal Weight | 8 lbs |
| Young's Modulus | 30 E + 6 psi |
| Poisson's Ratio | 0.24 |

3.2.3 Marmon Clamp

The Marmon clamp used for TerraSat will be the same size as the Defense Systems, Inc. bus discussed in Section 4. LEOSat has made one design change to the MDI Marmon clamp, which is

shown in Figure 3.2-3. In order to reduce space debris, LEOSat has added a 100 lb-test monofilament line to the Marmon clamp. This monofilament line will allow the Marmon clamp to release the satellite, but will hold the clamp together to avoid pieces flying into space. The lines will be held on the clamp with high temperature epoxy, and will be placed in several locations on the clamp for redundancy. The weight of this monofilament line is negligible. All other aspects of the Marmon clamp will be the same as the design by MDI.

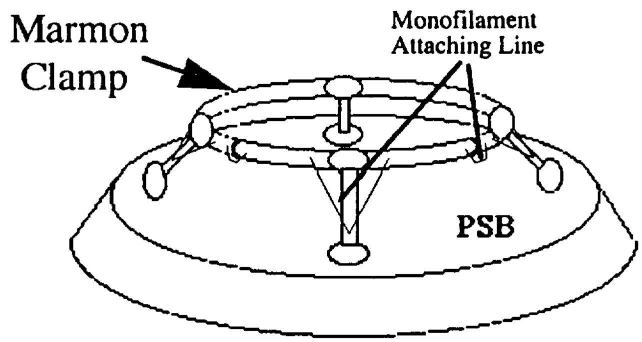


**Figure 3.2-3. PSB with modified Marmon clamp**

## 3.3 Orbit Analysis

Since TerraSat will provide data similar to that of the LANDSAT series of satellites, LEOSat has decided to use similar orbital parameters. LANDSAT 5 has an eccentricity of 0, an inclination of 98.22° and a semi-major axis of 7083 km (altitude = 705 km). Another requirement was that the satellite be in a sun-synchronous orbit and remain out of the shadow of the Earth. To determine which injection stage will be needed to achieve similar orbital elements, two TK! Solver routines were used. The first one calculated a sun-synchronous orbit altitude as a function of inclination. The second was a routine written by Minotaur Designs, Inc. in 1993 that models the performance of the Minuteman II booster. These routines are shown in Appendix F.

The MDI routine was run for an approximate mass of 127 lbs. for no injection stage and for Morton Thiokol STAR injection stages ranging from models 6 through 13F. The sun-synchronous orbit routine was run as well. These routines computed altitudes for inclinations ranging from 96° to 104°. The plot of the output of these routines is in Appendix F. As shown on the plot the altitude and inclination come closest to that of LANDSAT 5 by using a STAR 10 injection stage. The MDI

routine was then run again to find an exact mass which would give an exact sun-synchronous orbit. The orbit chosen has an inclination of 98.2 degrees, a semi-major axis of 7080 km, and an altitude of 702 km.. To achieve this inclination, the satellite must be launched from Vandenberg AFB and a Star 10 injection stage must be used. The chosen orbit resulted in a final mass for TerraSat of 145 kg. This extra 18 kg of mass may be accounted for by error margin, ballast mass, and the extra cryocoolers needed for the two infrared sensors. This orbit results in a sun-synchronous orbit that is out of the shadow of the Earth and is very similar to the orbit of LANDSAT 5.

## 3.4 GNC/Communications

### 3.4.1 Attitude Determination and Control

The specifications for Attitude Determination and Control Systems are listed in Table 3.4-1. Since TerraSat is a Earth remote sensing satellite, it must remain pointed at the Earth at all times. As a result, the DSI bus housing the TerraSat payload will be nadir pointing and three-axis stabilized, with a pointing error of 0.03 degrees. The High-Resolution UV/Visible Camera has the highest resolution of all the sensors with a field of view to be 0.3 x 0.4 degrees. For a remote sensing satellite, a pointing error of 10 to 20 percent of the sensor's field of view is usually recommended. To monitor these small angular rates of change, two star trackers, a three-axis magnetometer, and a horizon scanner will be used. In order to maintain TerraSat's altitude, small frequent maneuvers will be required to counteract orbital perturbations due to such effects as atmospheric drag and solar radiation pressure. LEOSat will use two reaction wheels and XYZ torque coils to perform such maneuvers.

**Table 3.4-1. Attitude Determination and Control**

| | |
|---|---|
| Design Approach | 3-Axis Stabilized |
| Accuracy | 0.1° |
| Reference | Nadir Pointing |
| Sensors | 2 Star Trackers<br>3-Axis Magnetometer<br>Horizon Scanner |
| Controllers | 2 Reaction Wheels<br>XYZ Torque Coils |

### 3.4.2 Communications

TerraSat will require a transmitter and a receiver to communicate with the ground stations, discussed in Section 3.6. Since uplink commands to TerraSat will require a modulation rate of no more than 9.6 kilobits/sec, a UHF-band command uplink will be used. However, due to the large amount of data to be transmitted to the ground station by the satellite, an S-band downlink with a modulation rate of at least 1 Megabit/sec will be required. To provide the capability of continuous imaging without the requirement of numerous ground stations for continuous downlinking, 100 Megabytes of on-board memory will be required.

## 3.5 Power/Thermal Control

### 3.5.1 Power

The total power required by the sensors is approximately 80 W. This power will be supplied by three sets of deployable solar panels and each set will provide 30 W for a total available power of 90 W. Body mounted solar panels will also be used for redundancy. The body mounted solar panels are capable of providing up to 20 W of orbit average power.

### 3.5.2 Thermal Control

Two of the four sensors, the Short-Wave and Long-Wave IR sensors, have their own cryocoolers. The cryocoolers used are the Integral Sterling type Ricor K506B coolers. They have an average mean-time-to-failure of 4000 hours and will maintain the sensors at a temperature between 50 K and 77 K. A total of six of these coolers will be added on board for a total design operational time of three years. The other two sensors will be kept between 253 K and 283 K by using the spacecraft bus as a heat sink. Since the satellite will be in a solar-synchronous orbit, the sensors will need to be cooled rather than heated.

## 3.6 Ground Operations

The work done in the area of ground operations falls into three categories. The first is the ground stations that will be needed to receive the data and monitor the satellite through the life of the project. The second is the plan for how to process the data after is has been received. The third is

the plan for distributing the processed data to teachers and students to fulfill the end goal of student involvement in the project.

### 3.6.1 Ground Stations

The LANDSAT project has been taken over by a private company called the EOSAT Company. In order to have an efficient Earth surveying by LANDSAT 4 and 5, the company has cooperated with other companies all over the world. There are 16 ground stations serving the EOSAT company right now. Their locations are listed in Table 3.6-1.

LEOSat has analyzed the feasibility of using existing LANDSAT ground stations to receive the data transmitted from TerraSat and has found this approach unfeasible. However, if LEOSat Industries can build a ground station modeled on the current EOSAT ground stations , it can begin an operation similar to that of EOSAT. It is estimated that construction of such a ground station will cost approximately $1 million. A good candidate for a model is the EOSAT ground station in Norman, Oklahoma which has a 10 meter and an 11 meter receiving dish. The ground station requires six persons per shift with two shifts per day. Personnel costs and operational costs for this ground station are approximately $0.5 million per year each. The data that is received by the ground station is stored on high density tapes by a VAX based computer. These tapes will be shipped directly to LEOSat headquarters in Austin, Texas, where they will undergo their first evaluation for quality (line drops, pixel noise, etc.) and cloud cover and will then be distributed to image processing centers across the United States.

**Table 3.6-1. Locations of existing ground stations**

| Continent | City |
|---|---|
| North America | Prince Albert (Canada) |
| North America | Goddard Space Center (Maryland, USA) |
| North America | Norman (Oklahoma, USA) |
| South America | Cotopaxi (Ecuador) |
| South America | Cuiaba (Brazil) |
| Europe | Kiruna (Sweden) |
| Europe | Fucino (Italy) |
| Africa | Pretoria (South Africa) |
| Asia | Riyadh (Saudi Arabia) |
| Asia | Islamabad (Pakistan) |
| Asia | Shadnagar (India) |
| Asia | Beijing (China) |
| Asia | Bangkok (Thailand) |
| Asia | Jakarta (Indonesia) |
| Asia | Hatoyama (Japan) |
| Australia | Alice Springs (Australia) |

### 3.6.2 Processing the Data

The first part of exploiting the data received from TerraSat involves getting the information processed. The most efficient plan would be to use graduate students at universities across the country, since universities provide the benefits of low-cost but highly skilled labor, cutting edge computer facilities, and experienced guides in the professors. One university will be selected as the monitor for the operation and 9 other universities will be selected to participate in the program. Two graduate students at each university will be funded at an average rate of $33,000 per year for a total cost of $660,000. This figure includes the 33% benefits and 50% overhead that will be charged by each university to maintain the account and pay the students. The students will be required to process a minimum number of images each year, which will be decided upon at a later date. Each graduate student will be required to write a minimum number of lesson plans from his or her newly processed images and forward these lesson plans to the monitoring university for compilation into a

library that will be made available to teachers across the U.S. for free and to teachers in other countries for a small charge.

### 3.6.3 Distribution to Teachers and Students

This process also involves two parts. The first part is direct distribution of advertisement on the new educational materials that are available to science, physics, and math teachers at high school across the country. The U.S. Department of Education Office of Educational Research and Improvement estimates that there are approximately 32,000 public and private high schools in the continental United States. A direct mail-out advertising the educational materials available from LEOSat industries, the teacher-outreach offices at the NASA centers around the country, and the information available from other satellite companies who wish to be included in the advertisement, will cost approximately $2200 for bulk rate postage. 40,000 copies of a two-color, one page brochure will cost approximately $25,000. Once the advertising has been distributed, two people will be employed to run copies of the lesson plans and ship them to teachers that request them. The two yearly salaries will cost approximately $60,000. A black-and-white copier and a color copier will cost approximately $5000 and will probably have to be replaced each year. Supplies for the copier will be approximately $3000 per year. Thus the total direct-mail approach will cost a maximum of $96,000.

As a second phase, the graduate students working to process the images will be required to produce lesson plans from the images they are digitizing and send the plans to the project's headquarters. These lessons plans will be consolidated into a lesson package and advertised in the mailout. The plans will be sent to teachers free of charge as requested. The graduate students will also be required to distribute the lesson plans to the high schools in their towns and visit the classes of surrounding high schools who request the service at least four times per year. Incentive pay will be provided to graduate students who visit more than four classes per year.

The second part of the information distribution process is distributing the finished images to interested undergraduate and graduate students everywhere using Internet. By posting files of finished images in a directory on Internet, students everywhere can access the information and become aware of the power of using small satellites for Earth-sensing. Distributing files will require

a large computer system and technical assistance with the system, a router and TI cable for easy access to Internet, and a computer programmer to run the system, watch for problems, and keep the directory updated with the latest files. Table 3.6-2 shows the approximate cost of setting up such a system. The salary for the computer programmer is considered one third the yearly salary of a company employee who has other duties at the company. The total cost for the first year is $73,000 for the first year and $29,000 each subsequent year.

**Table 3.6-2. Cost of Internet Distribution System**

| Item | Approximate Cost |
|---|---|
| Computer | $40,000 |
| Router | 4,000 |
| TI cable ($1000/month) | 12,000 |
| Technical Assistance (year) | 2,000 |
| Computer Programmer | 15,000 |
| Total in first year: | $73,000 |

### 3.6.4 Total Cost of Ground Operations

The cost of ground operations for the TerraSat project is itemized in Table 3.6-3. The total cost for the first year is estimated at $2.7 million and $1.2 million for each subsequent year.

**Table 3.6-3. Cost of Ground Operations**

| Item | Approximate Cost |
|---|---|
| Ground Stations | $2,000,000 |
| 20 Graduate Students | $660,000 |
| Direct Mailout | 96,000 |
| Internet Distribution | 73,000 |
| Total in first year: | $2,729,000 |

## 4.0 SATELLITE BUS SELECTION

LEOSat Industries has selected a commercially available satellite bus to carry the project payload into orbit. The primary restrictions placed on the spacecraft bus are that it be compatible with the MMII launch vehicle. The launch vehicle shroud is 90 inches high, 48 inches in diameter at the bottom and 35 inches in diameter at the top with a total volume of 85 cubic feet. The launch vehicle also places a mass constraint on the satellite bus. The maximum allowable mass of the payload is about 1100 lb, depending on the launch site, orbit altitude, and inclination. Therefore buses with lower mass will be a great advantage.

The Standard Satellite-1 made by Defense Systems, Inc. is an 8-sided, modular satellite developed for medium size payloads that are designed to operate in low Earth orbit. The SS-1 is 30 inches in diameter and has a core module height of 16.5 inches. The payload is housed in a variable height payload module, and the modules are designed such that they may be stacked on top of or beneath the core module and several payloads may be stacked on top of each other. The maximum allowable payload mass is 400 lb. and the maximum total mass of the bus and payload is 625 lb. The bus provides up to 150 Watts of power and may be 3-axis or spin stabilized. Figure 4-1 shows a simple diagram of the DSI SS-1 satellite bus.

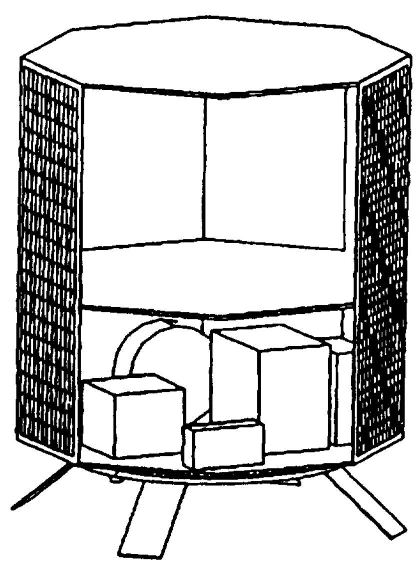

**Figure 4-1. DSI SS-1 Satellite Bus**

# 5.0 ESTIMATED MASS, VOLUME, COST, AND TIMELINE FOR TerraSat

## 5.1 Estimated Mass

Table 5.1-1 shows the estimated mass of TerraSat. The additional cryocoolers needed to extend the life of the sensors are included in the ballast and safety margin figure.

**Table 5.1-1 Estimated Mass**

| | |
|---|---|
| Dry mass of bus | 100 kg |
| Solar panels | 22 kg |
| Payload package | 5 kg |
| Ballast and safety margin | 18 kg |
| TOTAL | 145 kg |

## 5.2 Estimated Volume

Table 5.2-1 shows the estimated volume of TerraSat. The estimated volume of 1.8 $m^3$ is much less than the 3.1 $m^3$ volume available in the shroud. Figure 5.2-1 shows the inside of the shroud of the MMII launcher. This figure verifies that the satellite fits inside the shroud within a large margin.

**Table 5.2-1 Estimated Volume**

| | |
|---|---|
| Bus | 1.5 $m^3$ |
| Solar panels | 0.3 $m^3$ |
| TOTAL | 1.8 $m^3$ |

## 5.3 Estimated Cost

Table 5.3-1 shows the estimated cost of one satellite. The figure for the bus was obtained through a telephone interview with a representative of DSI. The figure for the payload package was obtained through a telephone interview with a representative of Ballistic Missile Defense Organization. The figure for the launch vehicle was obtained through an interview with Dr. W. T.

Fowler of the University of Texas at Austin. The figure for the injection stage was obtained from the MDI report from the Fall semester of 1993.

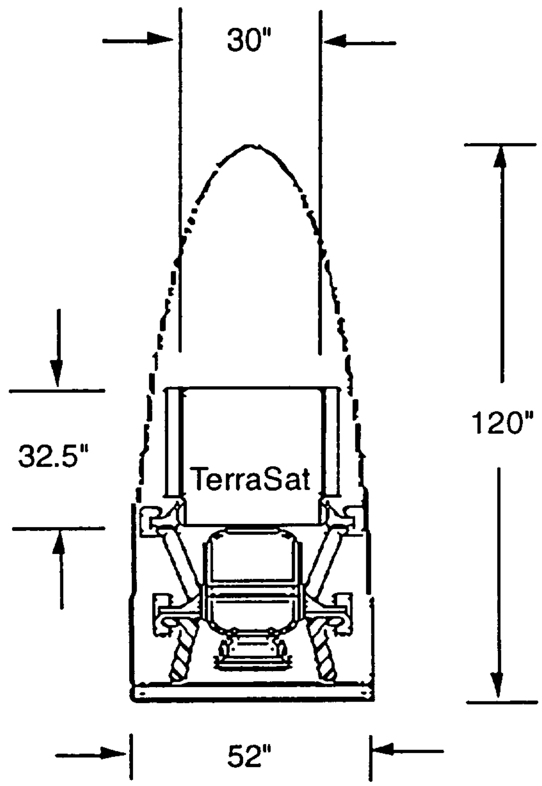


**Figure 5.2-1. Fit-check of Satellite in MMII Shroud**

**Table 5.1-1. Estimated Cost**

| | |
|---|---|
| Bus | $4.0 M |
| Payload Package | 1.7 M |
| MMII Launch Vehicle | 7.5 M |
| Injection Stage | 0.5 M |
| TOTAL | 13.7 M |

## 5.4 Estimated Timeline

Figure 5.4-1 shows the estimated timeline for the construction and launch of TerraSat. The total estimated time to launch is 13 months and the design life of each satellite is three years.

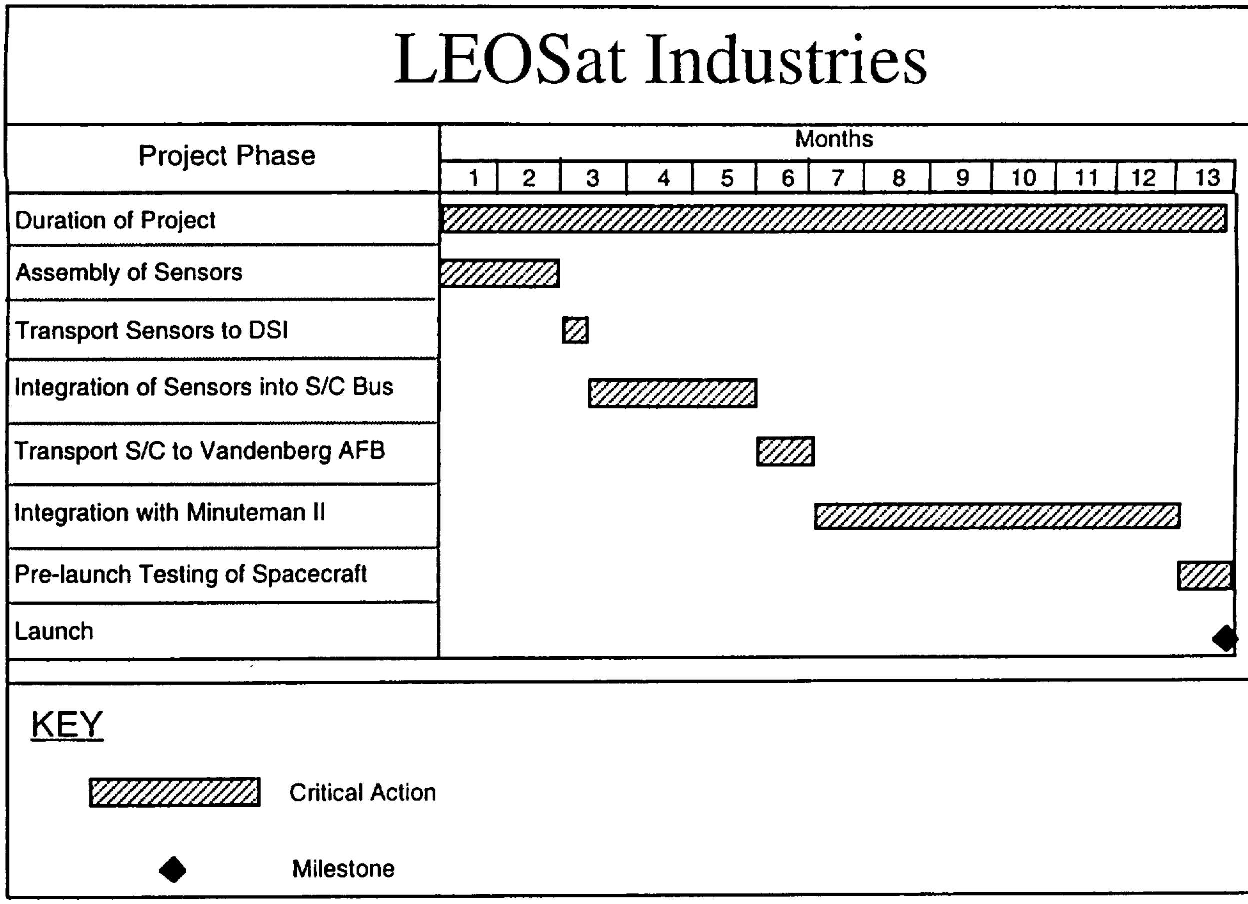


**Figure 5.4-1. Estimated Timeline**

## 6.0 COST AND TIMELINE FOR PROJECT

### 6.1 Cost of Project

#### 6.1.1 Personnel Cost

All of the LEOSat engineers are paid on a salary basis. Table 6.1-1 shows the salaries for the project manager and system engineers (salaries based on a 40 hour work week and a 52 week year).

**Table 6.1-1. LEOSat Salaries**

| Level | per hour | per year |
|---|---|---|
| Project Manager | $28.92 | $60,155 |
| System Engineers | $19.00 | $39,525 |

LEOSat's management structure has been simplified since the original cost proposal. One Design Manger position was changed to a System Engineer. For this reason, the actual personnel costs are much lower than had been originally projected (Table 6.1-2).

**Table 6.1-2. Actual Personnel Costs**

| PERSONNEL | wage | hrs | total |
|---|---|---|---|
| 1 Project Manager | $28.92 | 128.8 | $3724.90 |
| 5 System Engineers | $19.00 | 573.3 | $10892.7 |
| | **12 Week** | **Total:** | **$14617.60** |

The projected cost through week 12 was $18,648. LEOSat is well under budget by $4,030 on personnel costs due to the simplification of the management structure.

#### 6.1.2 Material Cost

The material and usage costs are estimated in Table 6.1-3. The computer expenses are based on 1994 computer usage fees in the Aerospace Engineering department at The University of Texas at

Austin. Telephone expenses are based on current long-distance rates. Project poster, model, photocopies, and transparency expenses are estimated from previous design projects.

**Table 6.1-3. Projected Material Costs**

| ITEM | COST |
|---|---|
| Computer Usage ($15 per team member) | $90.00 |
| Long Distance (10 hrs at $0.24 per minute) | $144.00 |
| Photocopies (1000 at $0.06 per copy) | $60.00 |
| View Graphs (100 at $0.50 per copy) | $50.00 |
| Documentation (project notebook) | $10.00 |
| ,Travel | $115.00 |
| Supplies (model and poster) | $100.00 |
| TOTAL | **$454.00** |

The total proposed cost of materials and usage fees was estimated at $454. Table 6.1-4 shows the actual costs.

**Table 6.1-4. Actual Material Costs**

| ITEM | COST |
|---|---|
| Computer Usage | $100.00 |
| Long Distance (10 hrs at $0.24 per minute) | $144.00 |
| Photocopies (1000 at $0.06 per copy) | $60.00 |
| View Graphs (100 at $0.50 per copy) | $50.00 |
| Documentation (project notebook) | $5.00 |
| Travel | $50.00 |
| Supplies (model and poster) | $49.00 |
| TOTAL: | **$458.00** |

LEOSat was over budget on material costs by only $4.

### 6.1.3 Consulting Cost

Technical consultants were utilized in the determination of a final design for this project. Consultants will be paid at a rate of $200.00 per hour. LEOSat estimated that a total of 15 hours of consulting time would accrue. Consulting costs were therefore projected to be $3,000. The actual cost is $9,000. LEOSat is significantly over budget on consulting costs due to unexpected difficulty in locating a satellite bus and miniaturized sensors.

### 6.1.4 Total Cost

The total estimated cost for this project consists of personnel costs, material costs, and consulting costs. The total estimated project cost was $28,375. The actual cost was $24072.

## 6.2 Design Strategy and Schedule

Figures 6.2-1 and 6.2-2 show the design scheduled that was followed and Figure 6.2-3 shows the approximate design plan that was decided upon before Preliminary Design Review 1.

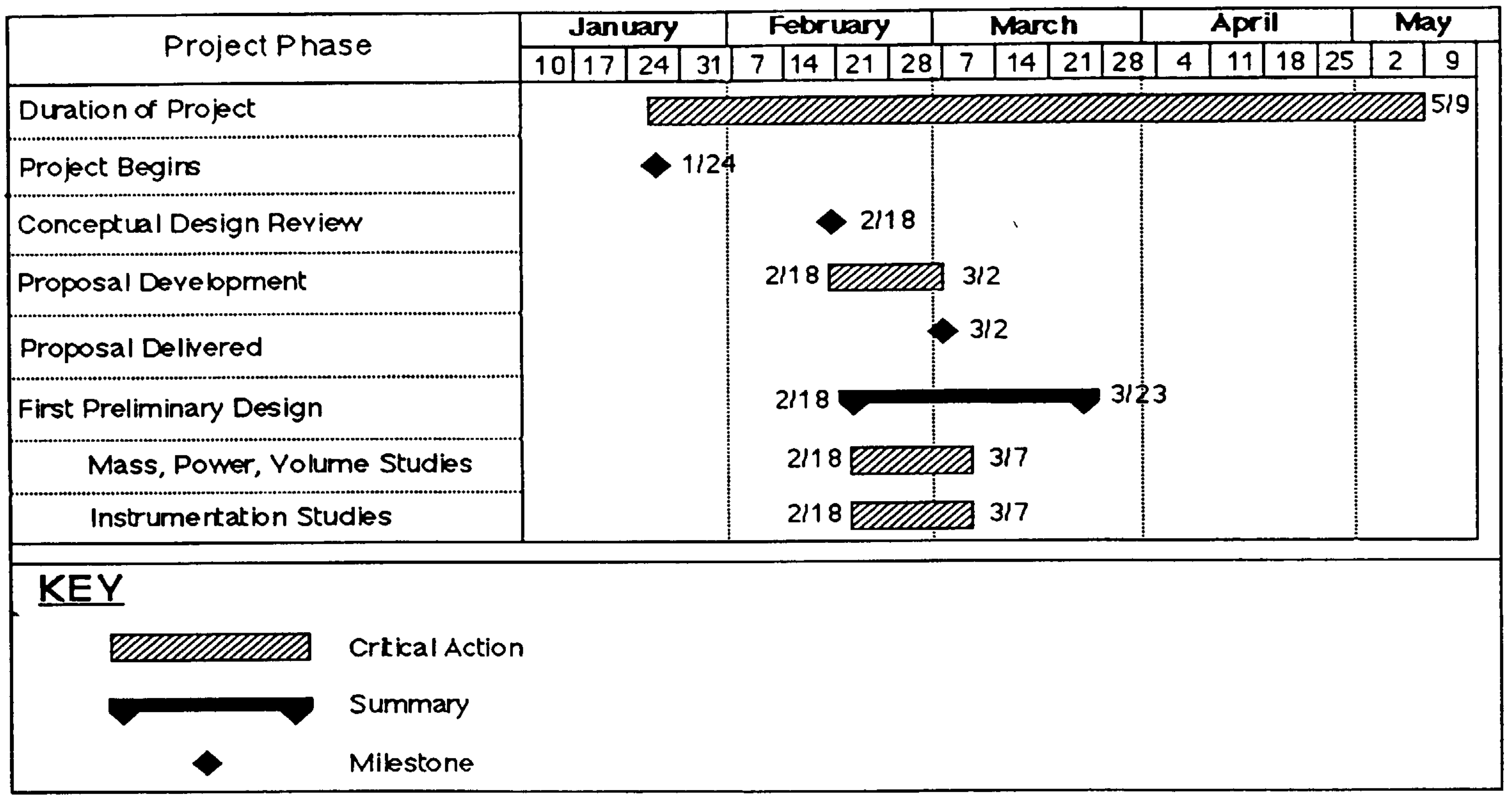


Figure 6.2-1 Project Schedule

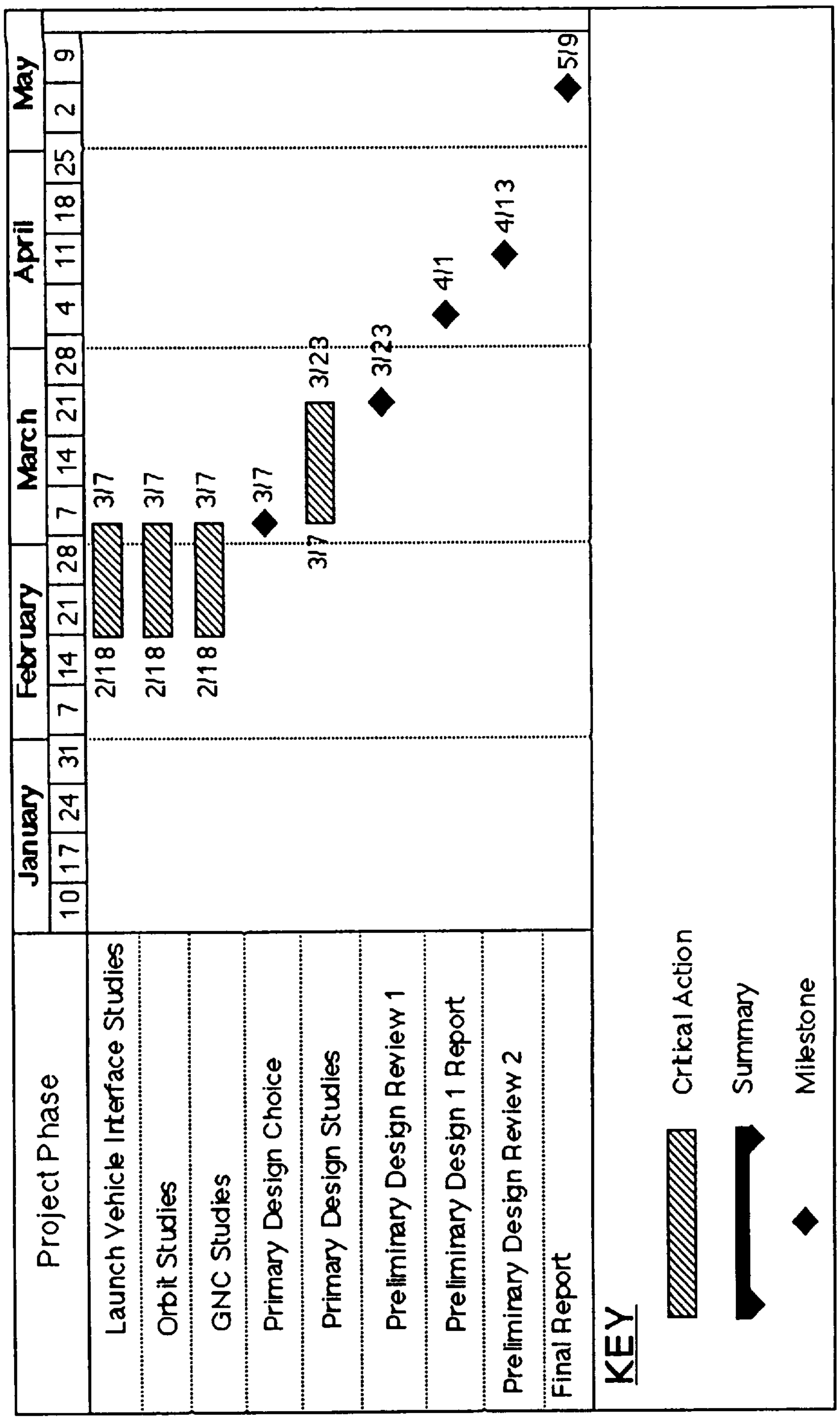


Figure 6.2-2. Project Schedule

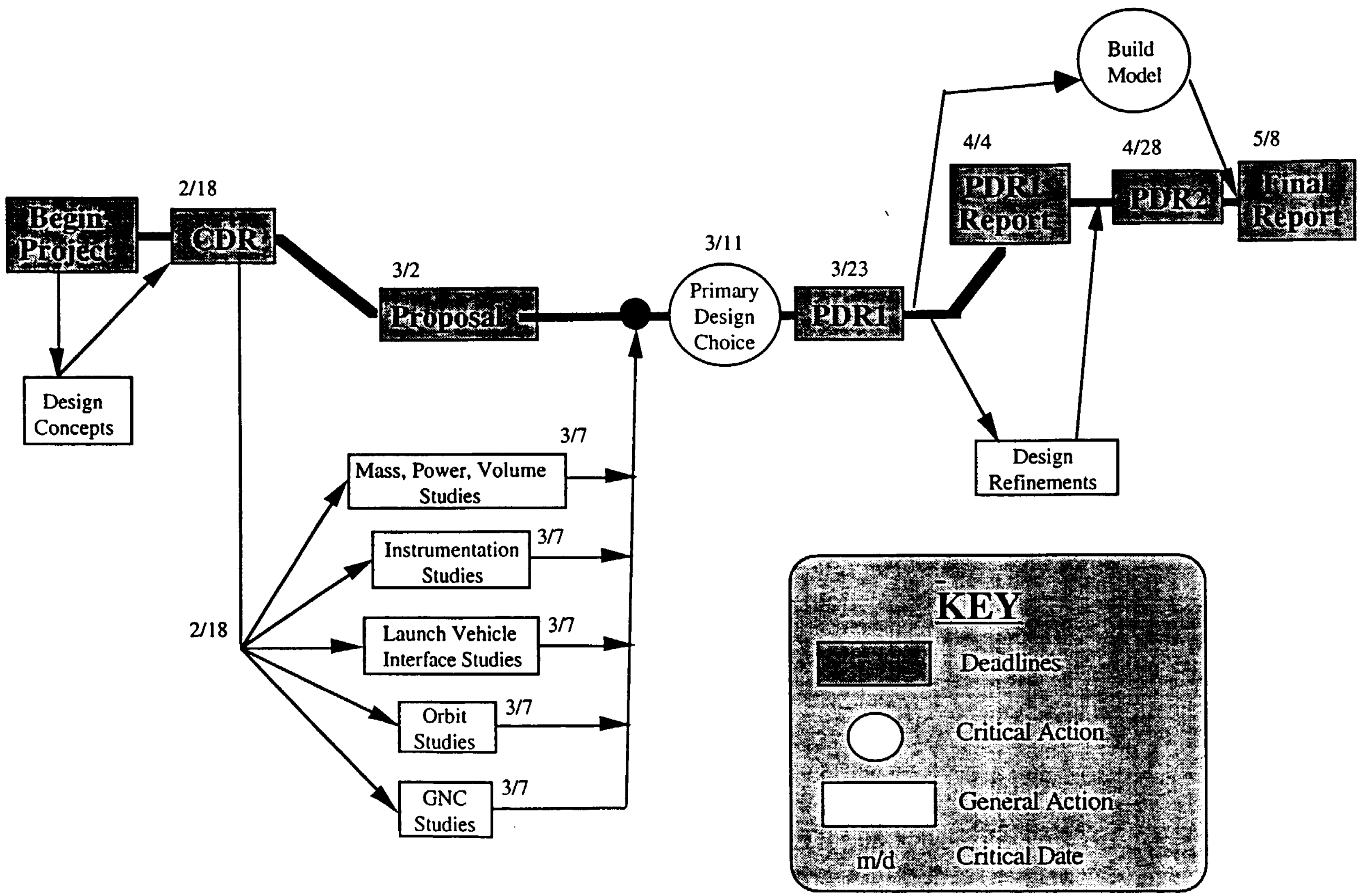


**Figure 6.2-3. Project Plan**

## 7.0 MANAGEMENT STRUCTURE

LEOSat Industries' design team is headed by a project manager. The primary responsibilities of the project manager is to insure the timely completion of the project by interfacing between the design team and the contractor. Administrative details are handled by the administrative officers and the project manager. For this design project, LEOSat has one design manager and five system engineers. Each subsystem is divided among the project manager and the system engineers as shown in Figure 7-1. Each subsystem group is responsible for the design of that system for both the primary and secondary satellite.

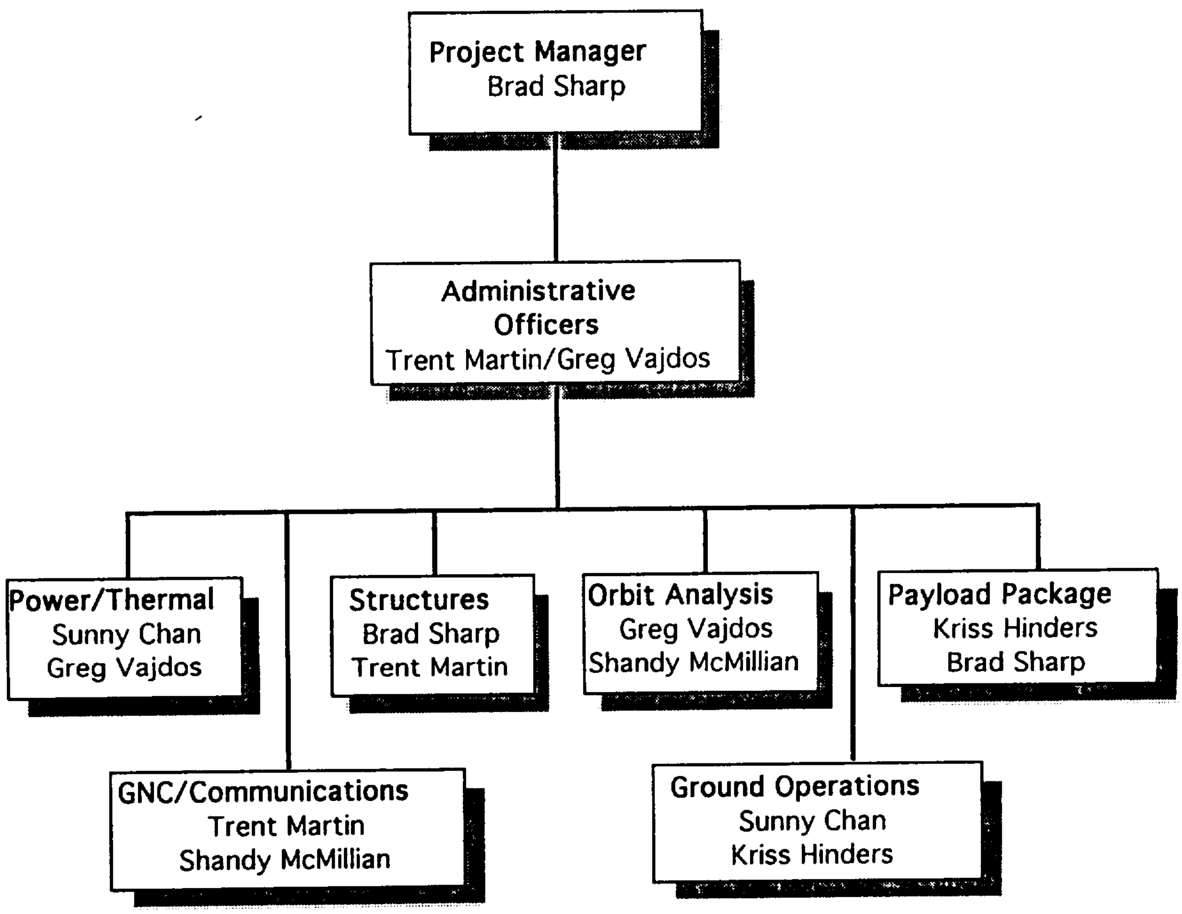


**Figure 7-1. Management Chart**

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## 8.8 Section 8

N/A

## Appendix A - Acronym List

| | |
|---|---|
| AFB | Air Force Base |
| CCD | Charge-Coupled Device |
| CGP | Crystal Growth Platform |
| CIB | Cosmic Infrared Background |
| CMB | Cosmic Microwave Background |
| COBE | Cosmic Background Explorer |
| COSPAS | Russian abbv. for Space System for Search of Vessels in Distress |
| DIRBE | Diffuse Infrared Background Experiment |
| DMR | Differential Microwave Radiometer |
| DSI | Defense Systems, Inc. |
| ELT | Emergency Locator Transmitter |
| EPIRB | Emergency Position-Indicating Radio Beacon |
| FIRAS | Far Infrared Absolute Spectrophotometer |
| GNC | Guidance, Navigation and Control |
| GPS | Global Positioning System |
| IR | Infrared |
| LUT | Local User Terminal |
| LWIR | Long Wave Infrared |
| MCC | Mission Control Center |
| MDI | Minotaur Designs, Inc. |
| MMII | Minuteman II |
| NIR | Near Infrared |
| OSC | Orbital Sciences Corp. |
| PLB | Personal Locator Beacon |
| RCC | Rescue Coordination Center |
| RFP | Request for Proposal |
| SARSAT | Search and Rescue Satellite Aided Tracking System |
| UV | Ultraviolet |

## Appendix B - Payload User's Guide

The following information was compiled by MDI for the benefit of satellite customers who are considering using the MDI missile as their launch vehicle. The main points discussed are the attainable orbits of launch system, the mechanical interfaces used to mount the satellite to the missile, electrical interfaces available for the payload, and launch system costs. This information can be found development

### B.1 Attainable Orbits

Figures B.1.1 and B.1.2 summarize the attainable orbits customers can expect from the MDI launch system. To meet a wide range of orbit needs, MDI can provide orbit inclinations between 29° and 57° by launching from Cape Canaveral and inclinations between 57° and 104° by launching from Vandenberg. The customer may use either a STAR 17A or STAR 27 injection stage if the orbit requirements demand better performance.

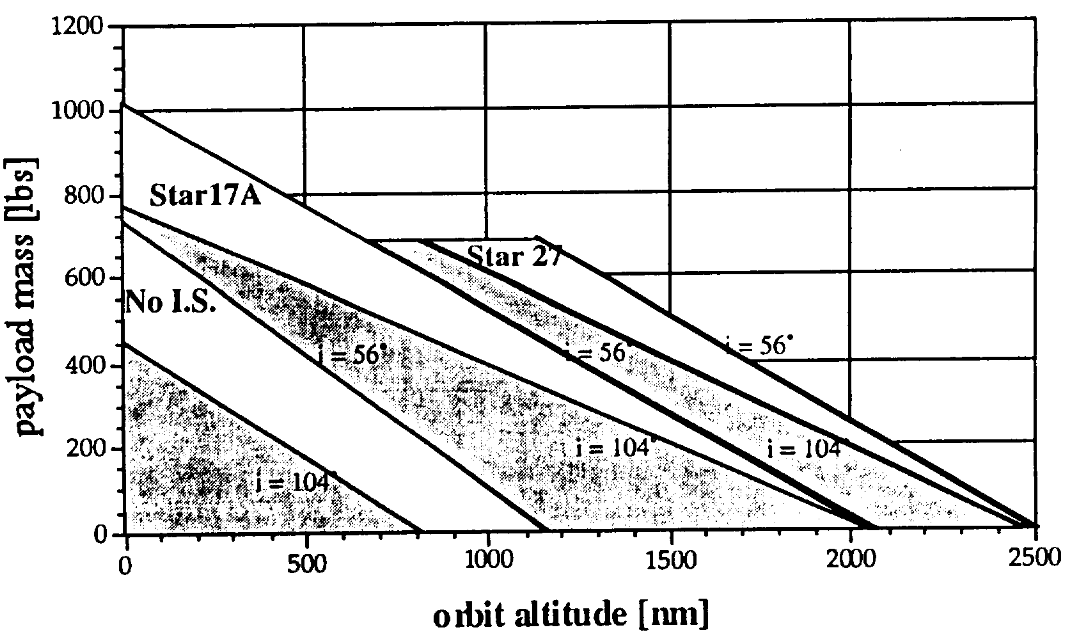



**Figure B.1.1 Launch Performance at VAFB Launch Site**

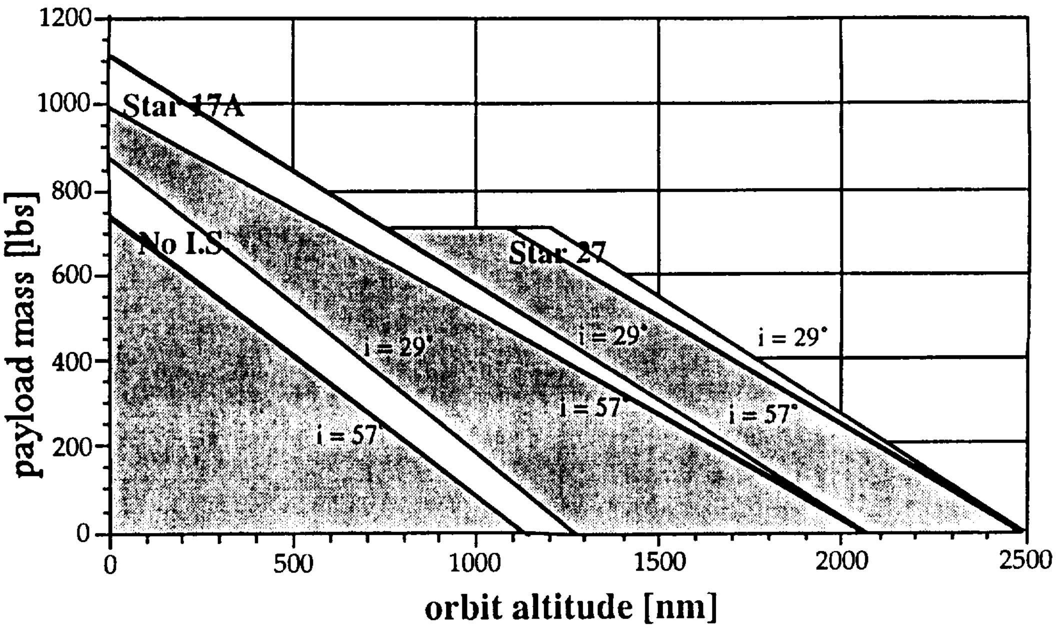



**Figure B.1.2. Launch Performance at Cape Canaveral Launch Site**

Like all launch systems, some orbit insertion error that the payload will need to tolerate. MDI expects orbit altitude errors of ± 15 nm and inclination errors of ± 0.5°.

## B.2 Mechanical Interfaces

Payload customers may connect directly to the missile bulkhead with use of the flexible MDI Marmon clamp, see Figure B.2.1. If the payload requires an injection stage, MDI will provide for an interface between the injection stage and payload as seen in Figure B.2.2.

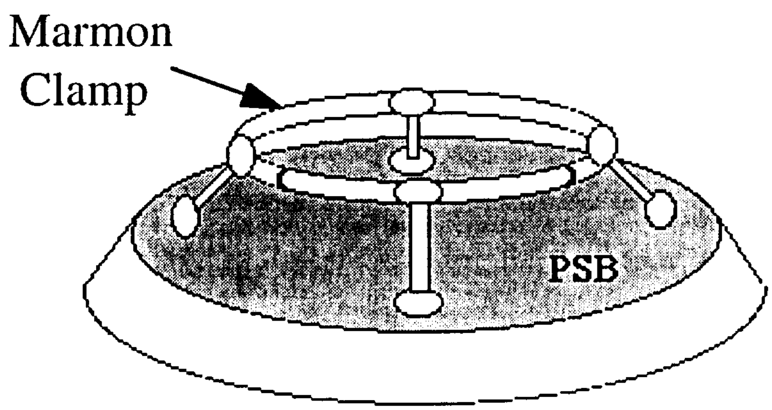



**Figure B.2.1. MDI Marmon Clamp**

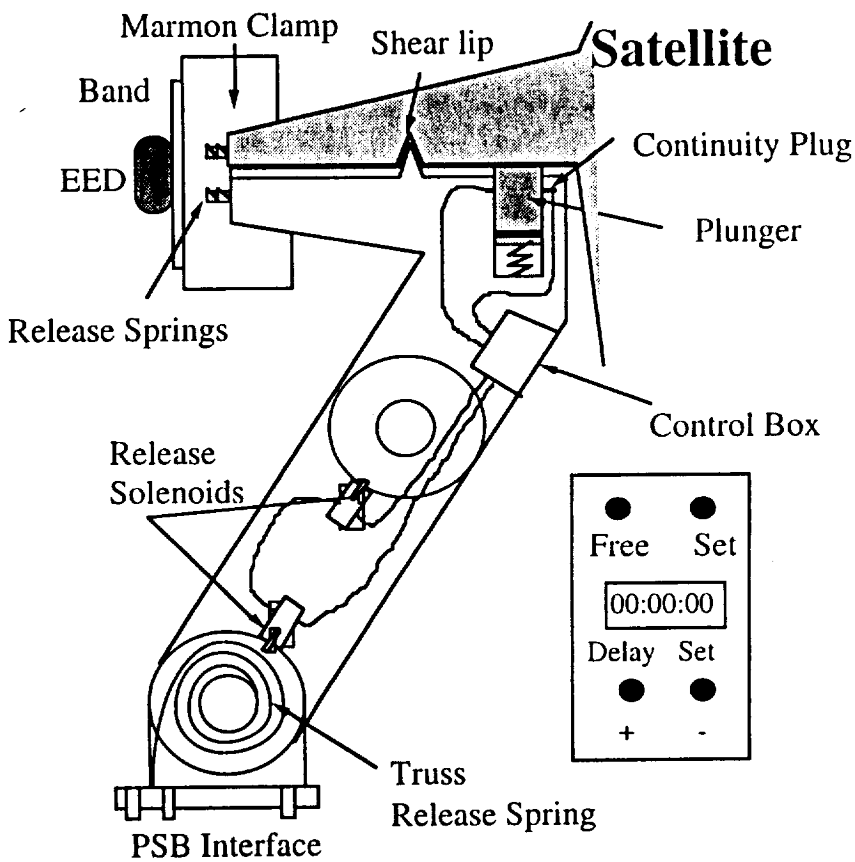



**Figure B.2.2 MDI Payload Attach Fitting**

B.3 Payload Support Bulkhead

The payload support bulkhead is identical to the MSLS payload support bulkhead with the exception of the holes drilled in the bulkhead. The MDI design relies on a quadrilateral truss design.

MDI's design requires the MSLS bulkhead be delivered without payload attach holes. The electrical interface is also different between designs. MSLS has three electrical connector groups. MDI will have only one. The PSB then has the same elemental and dimensional properties as the MSLS. The PSB dimensions are listed in Table B.3.1. Material properties are listed in Table B.3.2. The PSB is shown in Figure B.3.1.

**Table B.3.1 PSB dimensions**

| | |
|---|---|
| Height | 15 in. |
| Base Diameter | 52 in. |
| Bulkhead Diameter | 47.5 in. |
| Truss attach hole radius | 20 in. |
| Electrical hole radius | 20 in. |
| Electrical hole off axis | 45 deg. |

**Table B.3.2 PSB material properties**

| | |
|---|---|
| Material | Aluminum 7075 |
| Finish | Alodine 600 |
| Thickness | 2.0 inches |
| Weight | 25 lb. |
| Truss tension pullout capability | 15,000 lb. per truss |

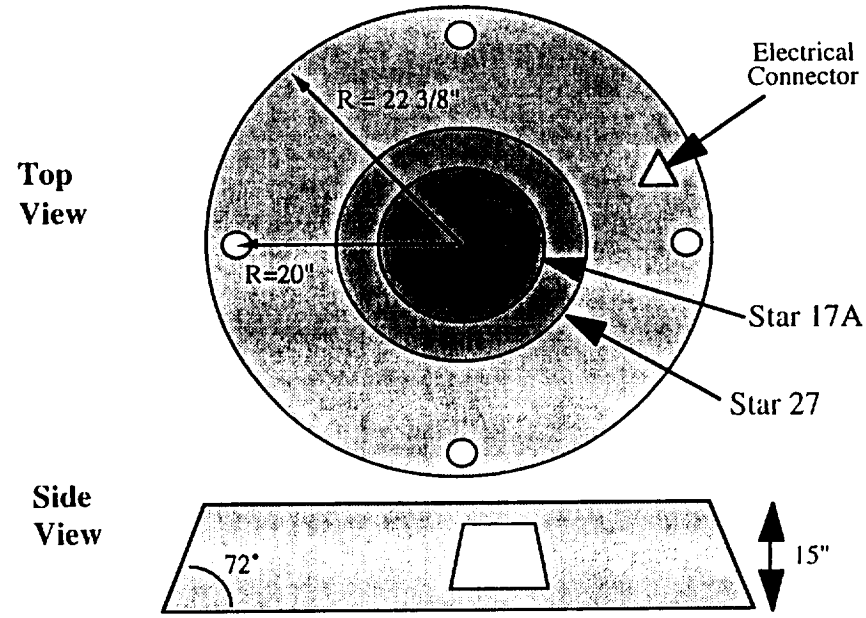



**Figure B.3.1 Payload Support Bulkhead**

B.4 Shroud

The shroud is made of a composite material that is RF transparent. The shroud is a secant ogive cylinder. Dimensions of the shroud are given in Table B.4.1. The shroud is shown in Figure B.4.1.

**Table B.4.1 Shroud dimensions**

| | |
|---|---|
| **Diameter** | 52 inches |
| **Height** | 120 inches |
| **Weight** | 214 lbs |

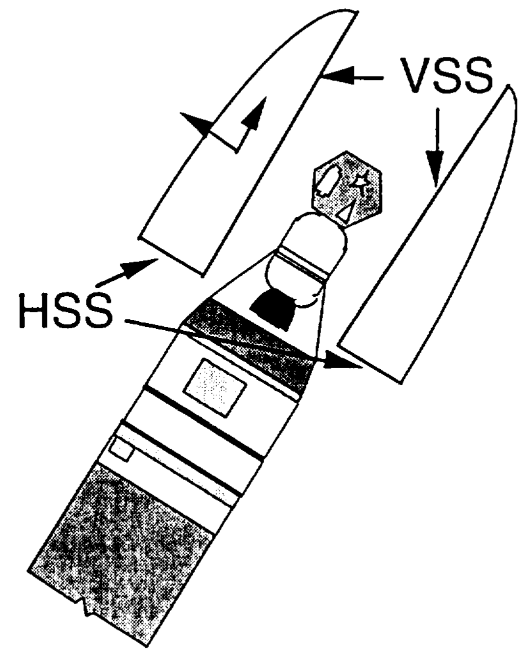



**Figure B.4.1 MDI shroud**

B.5 Launch Time and Costs

Once a customer decides to use the MDI missile as their launch system, they can expect a launch after 6 months and the following launch costs:

**Table B.5.1 Launch Costs**

| Launch Type | Cost |
|---|---|
| Regular Launch | $ 7,500,000 |
| Added STAR 17A | +$500,000 |
| Added STAR 27A | +$1,000,000 |

B.6 Expected Launch Environment

Figures B.6.1 thru B.6.4 show the anticipated launch environment that a potential satellite will be exposed to.

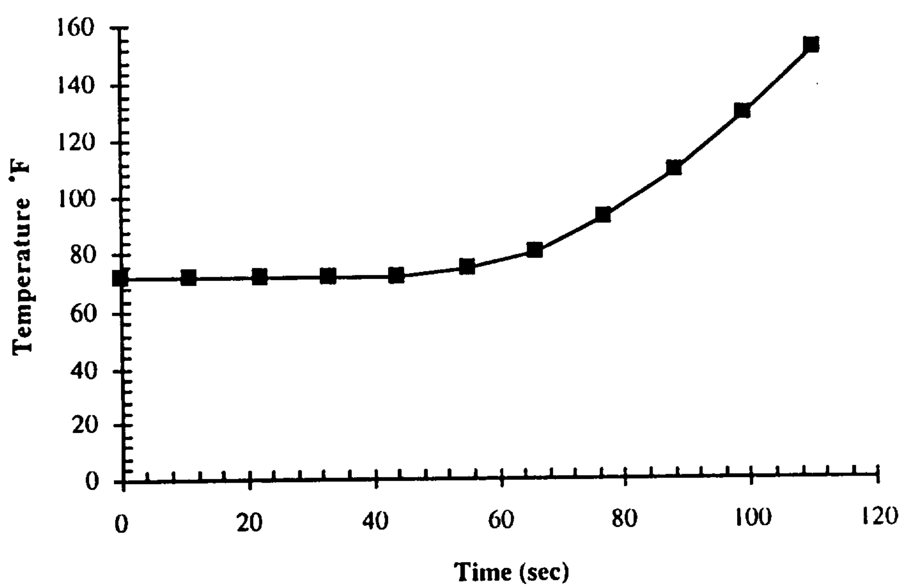



**Figure B.6.1 Thermal environment**

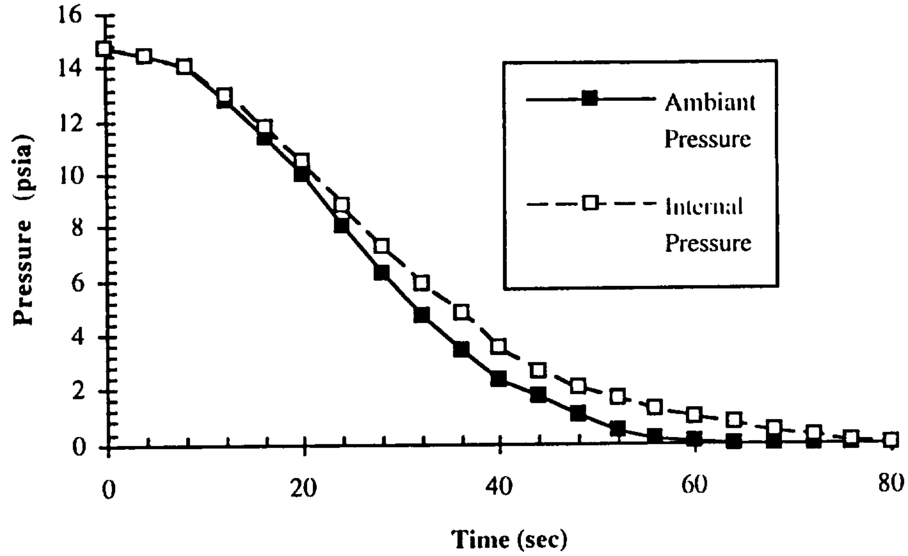



**Figure B.6.2 Pressure environment**

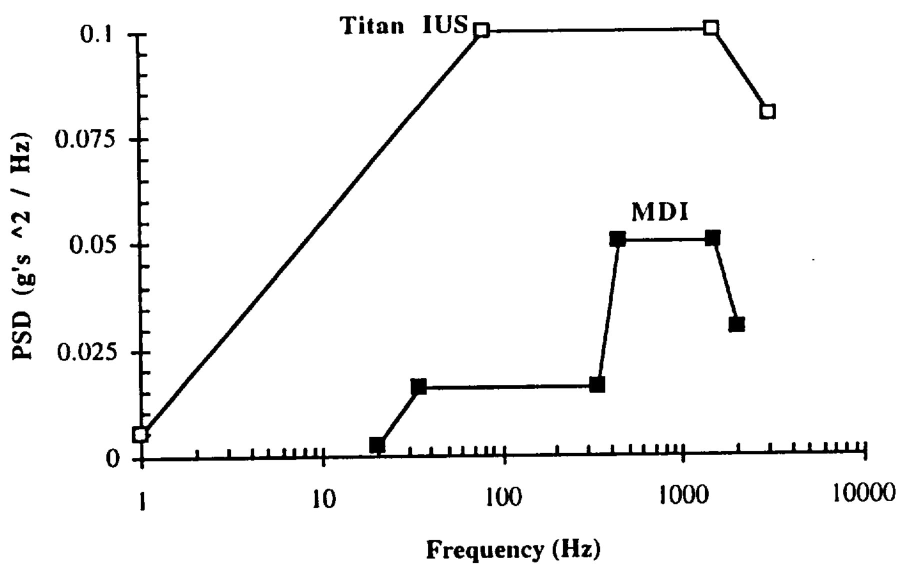



**Figure B.6.3 Random vibration environment**

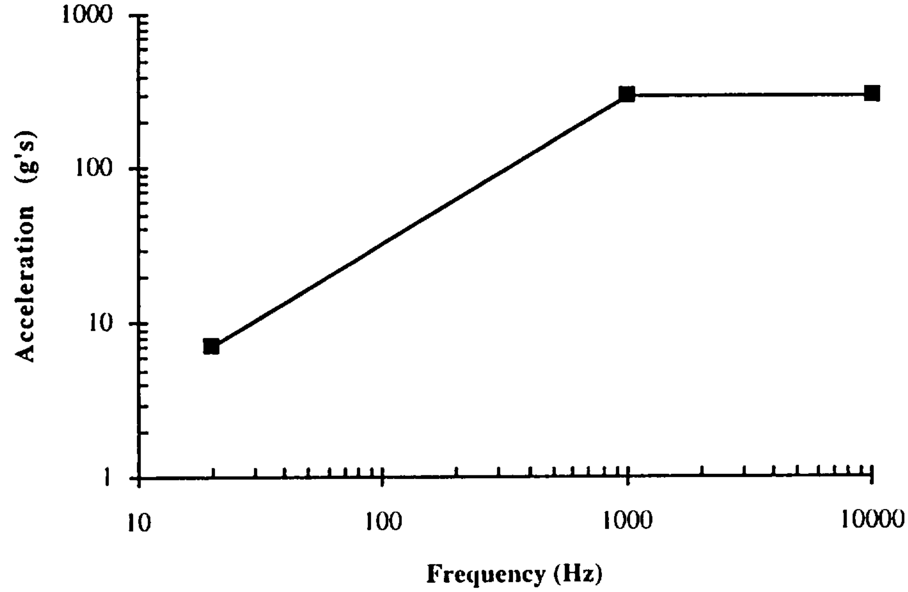



**Figure B.6.4 Shock environment**

## Appendix C - Secondary Design Project, COBE Jr.

### C.1 Background

In November 1989, NASA-Goddard Space Center used a Delta rocket to launch a science mission that was proposed as early as 1974, the Cosmic Background Explorer (COBE) satellite. The COBE used three sets of instruments, the Far Infrared Absolute Spectrophotometer (FIRAS), the Diffuse Infrared Background Experiment (DIRBE), and the Differential Microwave Radiometer (DMR), to make all-sky surveys in the millimeter, sub-millimeter, and infrared bands. The goals of the project were detecting and studying the Cosmic Infrared Background (CIB), and making detailed studies of the Cosmic Microwave Background (CMB). The smallest and cheapest experiment, the DMR, measured the differences in radiation from two points in the sky, thereby determining whether the radiation is isotropic. This new knowledge will answer some questions scientists have about the change that has occurred from the uniform distribution of mass that should have been present directly after the Big Bang to the non-uniform mass distribution of the universe today. All three instruments remained operational until NASA shut down the last of COBE's systems in February 1994.

Some of the theories about the Big Bang were considered proven after the data from the DMR was analyzed, while other theories were completely discarded. But in any important scientific experiment, the final proof does not come until the experiment has been successfully repeated, thus proving the method of obtaining the data and the instruments for obtaining the data. Because the data from the DMR experiment was only collected once, the COBE Jr. satellite will be used to repeat the DMR portion of the COBE experiments to verify the data and continue the research into cosmic background radiation.

### C.2 Project Summary

The COBE Jr. project will complete the experiment performed by the Differential Microwave Radiometer portion of the COBE satellite. One DMR at each of three wavelengths (31, 53, and 90 GHz) will measure the background radiation of the sky to check on the legitimacy and accuracy of

the data gathered by the COBE. The satellite will consist of the three DMR instruments mounted on a commercially-available satellite bus (see Section 4).

## C.3 Student Involvement

One of the requirements of the MMII small satellite project is to have a high level of student involvement to excite the next generation to the many possibilities and uses of space. The COBE Jr. has a fair amount of student involvement, although mostly at the higher levels of the education system. Undergraduate and graduate students could process the raw data that COBE Jr. obtains and produce all-sky maps of cosmic background radiation. By having several groups working with the data across the country, an independent verification of the methods used to reduce the COBE data could be obtained. Next, the finished data and maps could be released on an educational data base to allow graduate and undergraduate students to analyze the data, verify or refute existing theories on the Big Bang, and possibly develop new theories. The data could also be used by undergraduate and high school science teachers to introduce the idea of the Big Bang and its theories. It could also demonstrate how the scientific method is used to develop and prove scientific theories.

## C.4 Areas of Analysis

The following are areas that need to be analyzed to provide a detailed design for the COBE Jr. LEOSat Industries completed the first analysis, changes to the orbital parameters. However, there was not enough time to complete studies in the other areas. Future groups can use the information that follows as an outline for a future design project.

### C.4.1 Changes to the Orbital Parameters

Compromises between the three instruments on COBE led to an orbit design that called for the COBE to pass through the shadow of the Earth, which caused all the instruments to cool down. The instruments were very temperature sensitive and data gathered at temperatures other than the design temperature of ~140 K had to be discarded. The path through the shadow also required

substantial battery capacity for use when the solar panels were ineffective and larger solar panels to provide additional power for charging those batteries.

The trajectory for the COBE Jr. will be assigned with the principal goal of avoiding a path through the shadow of the Earth. Not only will there be no discarded data due to changes in instrument operating temperature, the capacity of both the batteries and the solar panels can be reduced. The trajectory analysis will also take into account the limitations on altitude and satellite mass placed on the satellite by the MMII booster. It has been determined that the launch must take place from Vandenberg AFB to provide the desired sun-synchronous orbit.

### C.4.2 Dicke Switches

A differential microwave radiometer is a device whose output voltage is proportional to the difference in power received by two horn antennas, shown in Figure 3-1. On the COBE, each DMR contained two independent radiometers, or channels, which operated at the same frequency. The output of each channel was proportional to the temperature difference of the regions of sky viewed by its horn pair plus an additive constant. The second channel allowed ground controllers to compare two sets of data at the same point to watch for failures and also provided a redundant system in case of failure. To obtain the temperature difference between the horn antennas, the experiment controller used a Dicke ferrite waveguide switch to connect the receiver input to one horn and then the other at a rate of 100 cycles per second. After some filtering, the difference signal that results was recorded every 0.5 seconds for telemetry to the ground.

The one problem with this system was that the Dicke switches that were used were very sensitive to magnetic fields. Their reaction characteristics varied as the satellite orbited around the Earth and as it rotated about its spin axis. This problem was fixed using the software onboard the COBE, but it meant thousands more lines of code. To save on programming costs and increase the accuracy and reliability of the COBE Jr., an analysis will be completed to find switches which are not as susceptible to magnetic fields but that are available at the lowest possible cost.

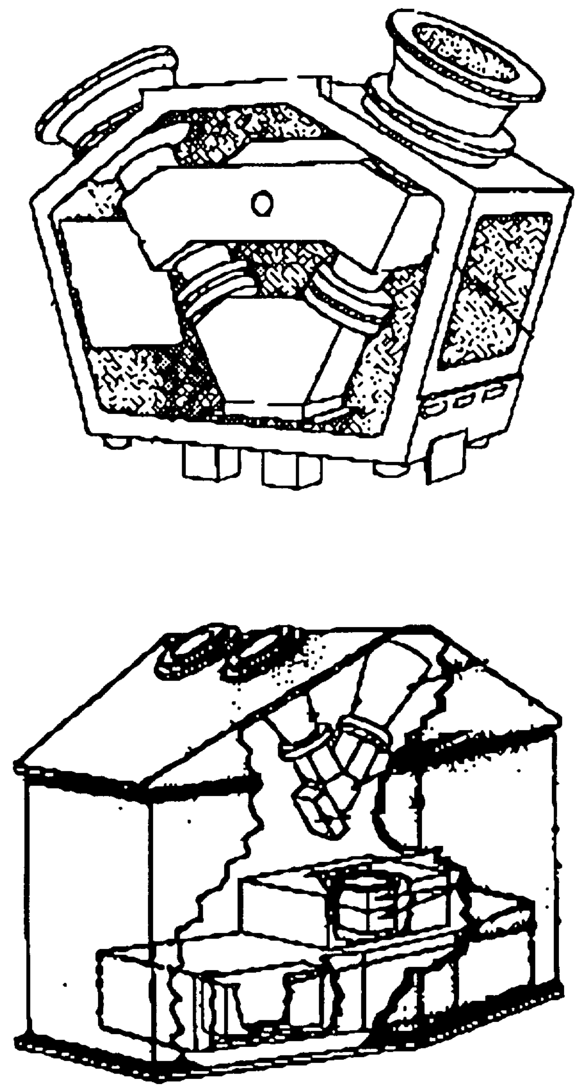


**Figure C-1. Typical horn antenna pair and internal configuration of a DMR.**

C.4.3 Receivers

The COBE DMRs used receivers that were designed and built in the mid-1980's. With advances in technology over the past few years, receivers now available are many times more sensitive and lighter than the ones used on the COBE. An analysis will be done to find the best possible receivers for the lowest cost and lightest weight.

C.4.4 Aperture change

Varying the aperture of the instrument changes both the angular scale and the weight of the instrument substantially. After a weight analysis was completed, NASA-Goddard selected an angular scale for the COBE antennas of 7°. For the COBE Jr., another analysis should be done to

determine the best angular scale that is achievable considering the MMII-imposed weight limitations.

### C.4.5 Earth/sun shield change

Ideal conditions for the Diffuse Infrared Background Experiment (DIRBE) included a completely unblocked view of the sky. However, ideal conditions for the DMR experiment dictated the inclusion of an Earth/sun shield to block the radiation given off by the Earth and the sun at critical times during the satellite's orbit. This Earth/sun shield blocked part of the view of the DIRBE, and thus another compromise had to be reached. The height of the Earth/sun shield had to allow the maximum possible range of view for the DIRBE while providing the maximum possible protection for the DMR experiment.

Since the COBE Jr. satellite will not face the same constraints as the COBE, the Earth/sun shield can be made as high as is deemed necessary with no regard to other experiments. The additional height will add volume and weight to the payload package and an analysis would have to be done to determine the maximum shield height that is still within weight and dimensional limits.

### C.4.6 Operating temperature change

The DMR instruments were operated at ~140 K. This temperature was decided upon after an analysis of the amount of weight needed by passive and active cooling systems to maintain various operating temperatures. The weight-temperature analysis had to include the weight of the other two experiments and the weight ceiling imposed on the satellite by the launch vehicle.

A target operating temperature of 70 K has been suggested for the instrument on the COBE Jr. An in-depth analysis could be done to determine the size and weight of the thermal control system that would be needed to maintain the DMR instruments at 70 K.

## C.5 Requirements for Secondary Project Design

The following sections detail the design specifics that were promised as stated in Section 5.3 of the Preliminary Design Review 1 report.

### C.5.1 Orbital Element Set

The following set of orbital elements was calculated using the TK! Solver program developed by MDI, discussed in Section 3.3, and shown in Appendix F. The program calculated the performance of the launch vehicle and injection stages to determine possible orbit elements for the weight and sun-synchronous requirement of the COBE Jr.

Table C.1 Orbital Element Set

| Orbital Element | Initial Design Value |
|---|---|
| Semi-major Axis | 7269 km |
| Eccentricity | 0.0 |
| Inclination | 99.0° |

### C.5.2 Commercial Bus Specifications

The bus will be required to lift a 125 kg payload (see C.5.4) to an altitude of 891 km (see section C.5.1). It will be required to supply approximately 150 W of power and provide approximately 1 Gbyte of data transfer per year. The data handling system will be required to transmit DMR and housekeeping data at approximately 280 bps so that 24 hours worth of data can be transmitted to the ground station in two minutes. The satellite will be spin-stabilized and spin at a rate of 0.8 rpm. DSI has estimated the mass and volume of a satellite bus fitting these specifications to be 122 kg and 1.8 $m^3$.

### C.5.3 Preferred launch sites

To achieve an inclination of 99.0° the COBE Jr. will have to be fired from Vandenberg AFB in California.

### C.5.4 Estimated mass and volume

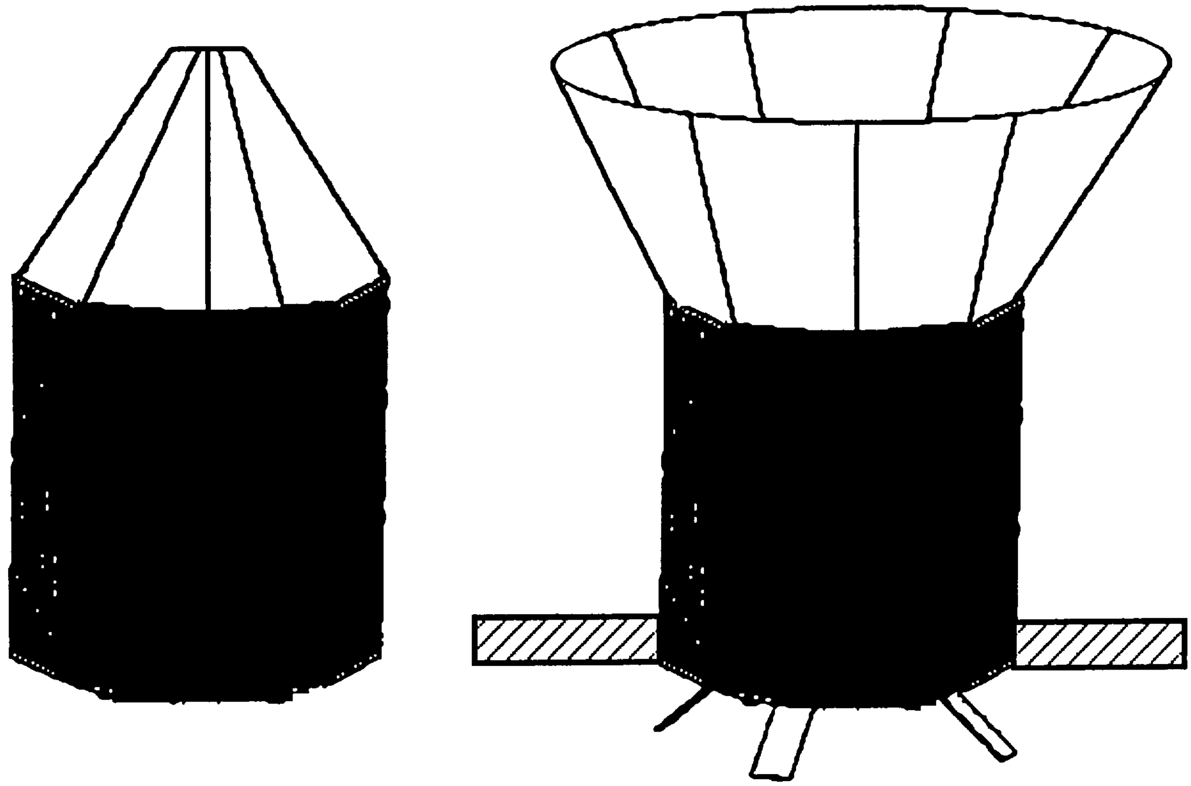
The total mass of the satellite will be the mass of the satellite bus, the mass of the payload package, and the mass of the Earth/sun shield. The mass of the satellite bus and solar panels has

been estimated by DSI to be approximately 122 kg. The contents of the instrument package payload will be very similar to the COBE Jr. The same number and type of instruments will be required for the COBE Jr. as flew on the COBE, although there may be some mass savings for instruments that have evolved into lighter, improved performance models over the past 5-10 years and there may be some mass costs for instruments that have evolved into higher performance models but at heavier weights. The total mass of the DMR portion of the payload package on COBE was 123.3 kg. The estimated mass of the payload package for COBE Jr. is 125 kg. The mass of the Earth/sun shield will depend on the optimal height decided upon after the analysis described in C.4.5. For this estimation, LEOSat will use the mass of the Earth/sun shield on COBE, which was approximately 15 kg. Thus the total estimated mass of the satellite is 262 kg.

The volume of the COBE will consist of the volume of the core module and payload module of the satellite bus and the volume of the Earth/sun shield. The volume of the satellite bus with solar panels deployed has been estimated by DSI to be 1.8 $m^3$. Again, the volume of COBE's Earth/sun shield in the stowed position will be used for this estimation and that volume is approximately 0.2 $m^3$. The estimated total volume for the satellite bus and stowed Earth/sun shield is approximately 2.0 $m^3$. With the Earth/sun shield and solar panels stowed, the satellite will be approximately 0.76 m in diameter and 1.6 m in length.

C.5.5 Sketch of secondary satellite

The following sketch of the COBE Jr. satellite, shown in Figure C.1 in both the stowed and deployed configurations, is only a sketch and is therefore not to scale. It should also be noted that the power requirements of the DMR instruments dictate the use of more than two solar panels. Only two panels were shown for simplicity. The other panels will be included in similar configurations around the satellite.



(a) Stowed/Launch Configuration (b) Deployed/Operational Configuration

**Figure C-2. Sketch of COBE Jr. in a) stowed and b) deployed configurations.**

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# Appendix D - Rejected Project, SOS

## D.1 Project Overview

Originally regarded as a likely candidate for LEOSat's primary satellite project, the SOS satellite was intended to demonstrate the ease and benefits of incorporating GPS (Global Positioning System) technology into the current satellite-aided search and rescue program for aircraft and marine vessels. The current system, COSPAS (a Russian abbreviation of Space System for Search of Vessels in Distress) and SARSAT (Search and Rescue Satellite-Aided Tracking System), relies on either an Emergency Locator Transmitter (ELT) installed on aircraft, or an Emergency Position Indicating Radio Beacon (EPIRB) installed on marine vessels, to transmit a distress signal at 121.5, 243, or 406 MHz frequencies when an emergency situation arises. This signal is eventually detected by a passing COSPAS or SARSAT satellite, which either immediately relays the 121.5 / 243 MHz signal to the ground or, in the case of the 406 MHz signal, processes it, computes the distress location, and continuously transmits the processed data and location to the ground. A Local User Terminal (LUT) then picks up the signal, computes the distress location if necessary, and passes the data to the search and rescue team via the Mission Control Center (MCC) and Rescue Coordination Center (RCC). However, despite the attractiveness of this system, it should be noted that the distress location is computed by way of Doppler shift techniques, yielding a position 5 km to 20 km away from the actual transmission site, and the whole process can take several hours.

SOS SAT, on the other hand, would have utilized modified ELTs and EPIRBs, which, when activated, would first use GPS to determine their latitude and longitude (within a few hundred meters), and then would transmit this information to the SOS satellite. The satellite would have relayed the "exact" location of the accident to the appropriate search and rescue team in the manner described above. Since the "exact" location would have been transmitted, the search time and the danger to both the rescuees and the rescuers would have been greatly reduced. In addition, the educational value of such a project would have been high, since students at undergraduate levels could have been involved in the design of experimental ELTs and EPIRBs, while high school students could have participated in the testing of the satellite. The satellite itself was also relatively

simple, primarily consisting of a commercial bus, a receiver/transmitter (a slightly modified SARSAT receiver/ transmitter), and support electronics.

However, after a few consultations with Ronald G. Wallace, Search and Rescue Mission Manager at Goddard Space Flight Center, LEOSat discovered that modifications to ELTs and EPIRBs alone would be sufficient to do the job of the SOS satellite. In response, LEOSat decided to relegate SOS SAT to a secondary payload on one or more of the other satellites being considered for design, and change its primary purpose to one of personnel, not aircraft or marine vessel, rescue. But again, additional research proved that this project was also unneeded. Discussions with Dave Affens, also of Goddard Space Flight Center, revealed that such a program can be handled by COSPAS / SARSAT, and will be carried out in less than three months by NASA, by mere modification of Personal Locator Beacons (PLBs). Thus this candidate has been rejected.

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## Appendix E - Rejected Project, Crystal Growth Platform

The second rejected project was the Crystal Growth Platform (CGP). This satellite was intended to produce high quality crystals at low price for use in industry and research. The market for such crystals would be fairly extensive, especially if a system could be devised to produce crystals expensively. For example, crystals like Gallium Arsenide (GeAs) are very important to the electronics industry because they can be used to produce devices which run more quickly than the devices currently using silicon crystals. Another example is Silver Chloride (AgCl ) has important applications in optical devices.

Although the existence of the Crystal Growth Apparatus developed and used by NASA made this project appear feasible, further research turned up many difficulties in this project. First, important factors in the growth of high quality crystals are temperature and pressure and maintaining the experimental environment of the satellite at extremely high pressures and high temperatures would be very difficult. Second, the technology of automated crystal growing apparatus is not yet mature, especially for a large-scale production effort. The crystal growth apparatus currently available requires almost constant supervision by astronauts and could not be used on a small satellite, thus requiring the development of an entirely new apparatus that could adapt to automation. Third, the shroud of the MMII cannot accommodate the COMET, the currently available reentry system. Also, COMET exceeds the weight budget of the launcher, which is expected to be lower than 680 lb. But even ignoring the size and weight of the COMET, the landing load of the recovery system is still much too high at 10g. Fourth, the goal of the Crystal Growth Platform is to provide more crystal for industry. The student involvement criteria cannot be met without addition of a secondary payload, such as computer imaging equipment to observe the crystal growth, which would increase the cost, complexity, and payload weight. In conclusion, the Crystal Growth Platform was rejected.

# Appendix F - TK! Solver Routines

## MDI Booster Performance TK! Solver Variable Sheet

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| | 6378.137 | re | | km | radius of Earth |
| | 7.2821E-5 | we | | rad/s | rotation rate of Earth |
| | .00981 | g | | km/s^2 | gravity const. at Earth surface |
| | 398600.44 | mu | | km3/s2 | gravity parameter |
| | 272 | mwaf | | kg | combined mass of wafers |
| | 109 | mshr | | kg | mass of shroud |
| | 20786 | mIp | | kg | stage I prop. mass |
| | 2129 | mId | | kg | stage I dry mass |
| | 6237 | mIIp | | kg | stage II prop. mass |
| | 797 | mIId | | kg | stage II dry mass |
| | 3313 | mIIIp | | kg | stage III prop. mass |
| | 338 | mIIId | | kg | stage III dry mass |
| | 6.027 | mISp | | kg | injection stage prop. mass |
| | | pmf | .89128009 | | prop. mass fraction |
| | 4.102 | mISd | | kg | injection stage dry mass |
| | 268 | Isp1 | | s | specific impulse 1 |
| | 287 | Isp2 | | s | specific impulse 2 |
| | 285 | Isp3 | | s | specific impulse 3 |
| | 273 | Isp4 | | s | specific impulse I.S. |
| | 0 | dvster | | km/s | steering loss |
| | 0 | dvdrag | | km/s | drag loss |
| | 0 | dvgrav | | km/s | gravity loss |
| | 1.2708961 | dvatpr | | km/s | atm. press. loss |
| L | | h | | km | circular orbit altitude |
| | | rp | 6378.137 | km | perigee radius of x-fer orbit |
| | | ra | 8476.8072 | km | apogee radius of x-fer orbit |
| | | a | 7427.4721 | km | semi-major axis of transfer |
| | | rvb | 5257.6505 | km | radius of Vandenberg |
| | | vvb | .38286736 | km/s | velocity of Vandenberg |
| | | vp | 8.445349 | km/s | peri. vel. of x-fer orbit |
| | | va | 6.3544672 | km/s | apo. vel. of x-fer orbit |
| | | vins | 6.8572934 | km/s | insertion vel. for cir. orbit |
| | | dvlaunc | 8.1099709 | km/s | first speed dv. |
| | | dvins | .50282625 | km/s | second speed dv. |
| | 34.48 | lat | | deg | latitude of Vandenberg |
| L | 96 | i | | deg | orbit inclination |
| | | beta1 | -17.06637 | deg | inertial launch azimuth |
| | | beta2 | 197.06637 | deg | inertial launch azimuth |
| | | eta | 107.06637 | deg | local angle b/w vvb and vp |
| | | az1 | 199.51538 | deg | local launch azimuth (i>87) |
| | | az2 | 160.48462 | deg | local launch azimuth (i<87) |
| | | m0 | 34279.18 | kg | mass before burn 1 |
| | | m1 | 13493.18 | kg | mass after burn 1 |
| | | m2 | 11364.18 | kg | mass before burn 2 |
| | | m3 | 5018.18 | kg | mass after burn 2 |
| | | m4 | 4221.18 | kg | mass before burn 3 |
| | | m5 | 908.18 | kg | mass after burn 3 |
| | | m6 | 298.18 | kg | mass before burn 4 |
| | | m7 | 186.36 | kg | mass after burn 4 |
| | | mf | 172.72 | kg | mass at orbit ins. (mpay) |
| | | dv1 | 2.4512327 | km/s | first burn |
| | | dv2 | 2.3013623 | km/s | second burn |
| | | dv3 | 4.2956205 | km/s | third burn |
| | | dv4 | 1.2910428 | km/s | fourth burn |
| | | dvtot | 10.339258 | km/s | total burn |
| | 140 | mpay | | kg | mass of the payload |
| | | hnm | 1133.1913 | nm | orbit altitude |
| | | mlbs | 386.8928 | lbs | payload mass |

# MDI Booster Performance TK! Solver Rule Sheet

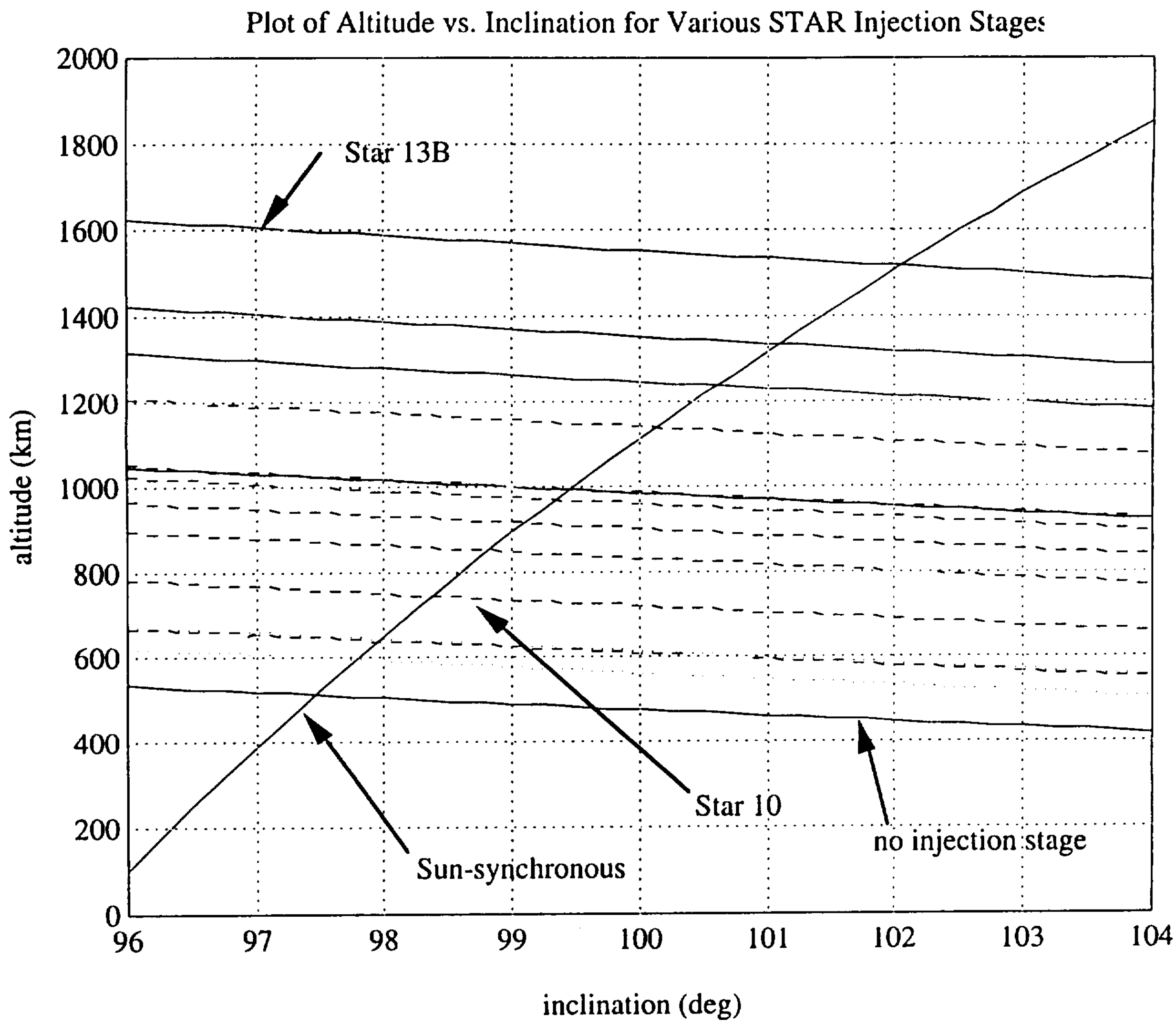
```
S       Rule
*       rp = re
*       ra = re + h
*       rvb = re * cosd(lat)
*       vvb = we * rvb
*       m0 = mIp + mId + mIIp + mshr + mIId + mIIIp + mIIId + mISp + mISd + mpay + mw
*       m1 = m0 - mIp
*       m2 = m1 - mId
*       m3 = m2 - mIIp - mshr
*       m4 = m3 - mIId
*       m5 = m4 - mIIIp
*       m6 = m5 - mIIId - mwaf
*       m7 = m6 - mISp
*       mf = m7 - mISd
*       pmf = mISp / (mISp + mISd)
*       dv1 = g * Isp1 * ln(m0/m1)
*       dv2 = g * Isp2 * ln(m2/m3)
*       dv3 = g * Isp3 * ln(m4/m5)
*       dv4 = g * Isp4 * ln(m6/m7)
*       dvtot = dv1 + dv2 + dv3 + dv4
*       a = (ra + rp)/2
*       vp = sqrt(mu * (2/rp - 1/a))
*       va = sqrt(mu * (2/ra - 1/a))
*       vins = sqrt(mu/ra)
*       dvins = vins - va
*       sind(beta1) = cosd(i)/cosd(lat)
*       beta2 = 180 - beta1
*       eta = beta2 - 90
*       dvlaunch = sqrt(vvb^2 + vp^2 - 2 *vvb*vp*cosd(eta))
*       sind(eta)/dvlaunch = sind(alpha1)/vp
*       alpha2 = 180 - alpha1
*       az1 = 270 - alpha1
*       az2 = 270 - alpha2
*       dvlaunch + dvins + dvdrag + dvgrav + dvster + dvatpr = dvtot
*       hnm = h/1.852
*       mlbs = mpay * 2.24
```

Sun-synchronous Orbit TK! Solver Variable Sheet

| St | Input | Name | Output | Unit | Comment |
|---|---|---|---|---|---|
| | 3443.9309 | Req | | nmi | Equatorial radius of Earth |
| | 62750.278 | mu | | nmi^3/s^2 | Earth's gravitational parameter |
| | 7.2921E-5 | We | | rad/s | Earth's rotation rate |
| | .0010826 | J2 | | | Oblateness Coefficient |
| | 1.991E-7 | Wj2 | | rad/s | Sunsynch Node Rotation due to J2 |
| L | | ALT | 3197.5731 | nmi | Altitude |
| L | | a | 6641.504 | nmi | Semi-major Axis of Orbit |
| | 0 | e | | | Eccentricity of Orbit |
| L | 170 | i | | deg | Inclination of Orbit |
| | | p | 6641.504 | nmi | Orbit Parameter |
| L | | r | 6641.504 | nmi | Radius of Orbit |
| L | | T | 13570.339 | sec | Period of Orbit |
| | | n | .00046301 | | Mean Motion of Orbit |
| LG | -56.54314 | Dlong | | deg | Change in Ascending Node per Rev |

Sun-synchronous Orbit TK! Solver Rule Sheet

| S | Rule |
| --- | --- |
| * | Wj2 = -3*(Req/p)^2*n*J2*cos(i)/2 |
| * | Dlong = (-We + Wj2)*T |
| * | a = r |
| * | p = a * (1 - e^2) |
| * | r = Req + ALT |
| * | T = 2*pi()/n |
| * | n = sqrt (mu/a^3)*(1+3*J2*(Req/a)^2*(3*(cos(i))^2-1)/(4*sqrt(1-e^2))) |



Plot of Altitude vs. Inclination for Various STAR Injection Stages
2000
1800
1600
1400
1200
1000
800
600
400
200
0
96
97
98
99
100
101
102
103
104
altitude (km)
inclination (deg)
Star 13B
Star 10
Sun-synchronous
no injection stage